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THEORETICAL INVESTIGATION OF ACCELERATION OF A TURBOJET ENGINE

Thesis by

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Thesis
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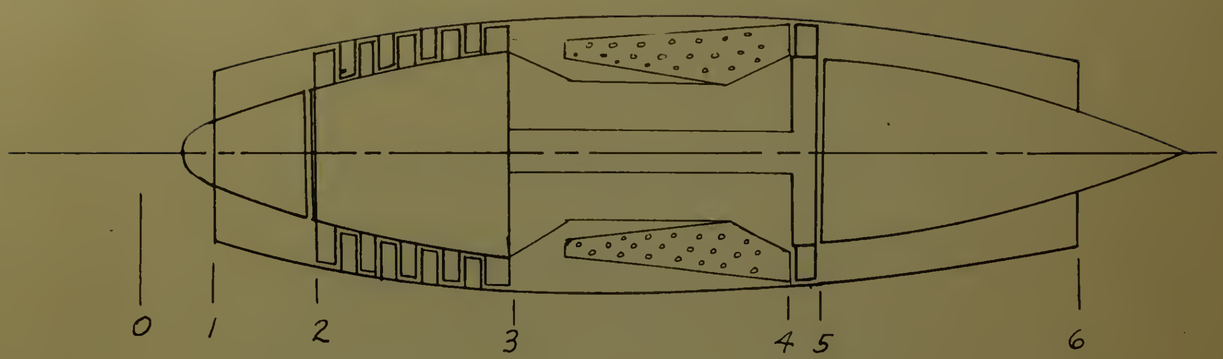
NOMENCLATURE



A_1, A_2, A_3, etc	Annular areas	ft^2
a_1, a_2, a_3, etc	Local sonic velocities	ft/sec
c_p, c_v	Specific heats	$\frac{ft - lb}{slug - {}^oR}$
D_c	Compressor inlet Mean diameter	ft
D_t	Turbine mean diameter	ft
F	Gross thrust	lb
g	Acceleration of gravity	ft/sec^2
I_p	Polar moment of inertia of rotor = 0.7675	$lb - ft - sec^2$
M	Mach number	
$\dot{m}$	Mass rate of fluid flow	slugs/sec
n	Rotational speed	RPM
$\dot{n}$	Rotor acceleration	Rev/sec^2

P_x	Free stream pressure at station "x"	lb/ft^2
P_{s_x}	Stagnation pressure at station "x"	lb/ft^2
R	Gas constant	$\frac{ft - lb}{slug - {}^\circ R}$
T_x	Free stream temperature at station "x"	$^\circ R$
T_{s_x}	Stagnation temperature at station "x"	$^\circ R$
V_x	Fluid velocity at station "x"	ft/sec
W_c	Compressor power	$\frac{ft - lb}{sec}$
W_t	Turbine power	$\frac{ft - lb}{sec}$
Γ_c	Compressor flow coefficient	$\Gamma_c = \frac{\dot{m}}{P_2 A_c}\sqrt{\frac{RT_2}{2}}$
Ω_c	Compressor power coefficient	$\Omega_c = \frac{W_c}{P_2 A_c a_2}$
σ_c	Compressor speed coefficient	$\sigma_c = \frac{D_c n}{a_2}$
Γ_t	Turbine flow coefficient	$\Gamma_t = \frac{\dot{m}}{P_4 A_t}\sqrt{\frac{RT_4}{2}}$
Ω_t	Turbine power coefficient	$\Omega_t = \frac{W_t}{P_4 A_t a_4}$
σ_t	Turbine speed coefficient	$\sigma_t = \frac{D_t n}{a_4}$
γ	Ratio of specific heats	C_p/C_v
η_d, η_b, η_n	Diffusor, burner, and nozzle efficiencies	
A_6/A_5	Tail area ratio	

SUMMARY

The failure of turbojet aircraft propulsion units to accelerate rapidly to high thrust operation when emergencies arise in slow speed flight has restricted their use in aircraft applications, and has also concentrated considerable attention upon their acceleration characteristics in an effort to produce better results. This thesis presents a method of computing the acceleration of a particular turbojet by making use of complete performance curves of the component parts of the turbojet.

The method presented here does not permit computation of the acceleration for a particular operating condition as determined by those variables usually considered independent; namely, (1) Flight Conditions of velocity, density, pressure, and temperature, (2) Engine rotor speed (3) Fuel rate of flow, and (4) Tail cone area ratio. Computation using these four independent variables was originally attempted in preparation of this thesis. However, extreme complication in the computations dictated that turbine inlet temperature and air mass rate of flow which are normally dependent variables, should be considered independent. Fuel rate of flow and tail cone area ratio are therefore considered dependent. Therefore, in order to match a particular operating condition, it is necessary to make a family of computations for various turbine inlet temperatures (constant for each set) over a range of assumed air mass flows.

Computations have been performed for the Westinghouse X19B axial flow turbojet to illustrate application of the method and to show qualitative and quantitative effects of variation of tail area ratio, at two different turbine inlet temperatures.

INTRODUCTION

During emergency conditions which arise in aircraft landing approach, a rapid increase in thrust is imperative. However, current turbojet engines are notoriously slow to accelerate from low to high thrust conditions. Although a large volume of information is available concerning equilibrium running conditions of turbojets, comparatively little has been published concerning acceleration. Accordingly the purpose of this paper is to develop a method of computation of acceleration, and of thrust during acceleration of a turbojet engine; and further, to ascertain qualitative effects on acceleration and thrust of variation of tail area ratio and turbine inlet temperature.

SCOPE

The basic method developed in this analysis is general and may be applied to any turbojet operating condition. However, this method does not encompass thrust augmentation devices such as afterburning. Application of the proposed method is contingent upon complete experimental performance data for the compressor, combustion chamber, and turbine, as separate units, and in addition upon knowledge of diffuser and nozzle efficiencies. The method is based upon the assumption that the steady state performance data can be applied instantaneously, even under non-stationary running conditions. Hence the accelerations are considered as "slow" changes, and although this assumption is probably valid, the final justification must come from comparison with tests.

The computations presented are restricted to a single flight condition. For this flight condition effects of variation of rotor speed from idle to military rating, and of tail area ratio from 0.8 to 2.0, are evaluated at two different turbine inlet temperatures.

PROCEDURE

In order to compute the acceleration of the rotor of the turbojet engine it is necessary to know both the power required to drive the compressor, W_c, and the power output of the turbine, W_t. The excess power, neglecting power required for the accessories, is then the power available for the acceleration of the rotor. In operation, the magnitude of these powers is determined by the independent variables: (1) Flight conditions (2) Rotor speed, n, (3) Fuel rate of flow and (4) Tail area ratio, A_6/A_5. However, for the purpose of calculation, the independent variables have been chosen to be: (1) Flight conditions (2) Air mass rate of flow (3) Turbine inlet temperature, T_4, and (4) Rotor speed, leaving fuel rate of flow and tail area ratio as dependent variables. This is completely explained in the discussion.

The first step used in calculating W_c and W_t was to find the entrance conditions of the compressor. Flight velocity of the engine was assumed to be 100 mph. Sea level standard conditions of density, pressure and temperature were assumed. Then assuming isentropic flow through the diffuser, a plot of M_2, T_2, and P_2, versus air mass flow, $\dot{m}$, was made by the use of a Mollier diagram. (See Figure 11) A particular rotor speed was then assumed. Values of mass flow from the lower limit of the compressor stall to the upper limit of the critical flow were chosen. The compressor flow parameter, Γ_c and speed parameter, σ_c were calculated for each mass flow, and then the compressor pressure ratio P_3/P_2, the compressor power coefficient

Ω_c, and the compressor temperature ratio, T_3/T_2, were found from Figures 13, 14, and 15. With these values it was possible to determine:

a) $P_3 = P_2\left(\frac{P_3}{P_2}\right)$

b) $T_3 = T_2\left(\frac{T_3}{T_2}\right)$

c) $W_c = \Omega_c P_2 A_c a_2$

P_4 was assumed to be 0.98 P_3. (See Figure 12). T_4 was then taken as the value desired: (2000°R for one case and 2200°R for the second case). From these data it was possible to compute

$$\sigma_t = \frac{D_t n}{a_4}$$

$$\Gamma_t = \frac{\dot{m}}{P_4 A_t}\sqrt{\frac{R T_4}{2}}$$

The turbine performance charts, Figures 16, 17, 18, and 19 were entered and from them was obtained:

a) $\frac{P_5}{P_4}$

b) Ω_t

c) $\frac{T_5}{T_4}$

d) M_5

From these the following were computed:

a) $P_5 = P_4\left(\frac{P_5}{P_4}\right)$

b) $W_t = \Omega_t P_4 A_t a_4$

c) $T_5 = T_4\left(\frac{T_5}{T_4}\right)$

d) $\frac{P_6}{P_5} = \frac{P_0}{P_5}$

(Unless P_{55}/P_0 exceeds the critical-then see Appendix II)

With this information the nozzle chart Figure 20 was entered and M_6, T_6/T_5, and A_6/A_5 were determined. From these data

$$T_6 = T_5\left(\frac{T_6}{T_5}\right)$$

and

$$V_6 = M_6\sqrt{\gamma R T_6}$$

The acceleration of the turbojet rotor was then computed as

$$\dot{n} = \frac{W_t - W_c}{4\pi^2 n I_p}$$

and the thrust

$$F = \dot{m}\left(V_6 - V_0\right)$$

for values of M_6 less than one. When P_5/P_6 exceeds the critical pressure ratio then the pressure at the nozzle outlet, P_6, exceeds atmospheric pressure, P_0, and a pressure term in amount $A_6\left(P_6 - P_0\right)$ is added to the thrust. Under this condition, $M_6 = 1.0$, and the expression for the thrust becomes:

$$F = \dot{m}\left(a_6 - V_0\right) + A_6\left(P_6 - P_0\right)$$

Where P_6 is determined from the known magnitude of P_5 and from that pressure ratio, P_5/P_6, which makes the value of M_6 equal to unity.

From the plots of the data obtained from the above calculations (See Figures 1, 2, 4, and 5) it was then possible to compute the acceleration time for the rotor by taking time increments of the order of a half a second and making the computation in a step by step process.

REFERENCES

1. "Jet Propulsion", A reference text prepared by the staffs of the Guggenheim Aeronautical Laboratory and the Jet Propulsion Laboratory, GALCIT, California Institute of Technology, for the Air Technical Service Command. (Restricted)

2. Westinghouse Aviation Gas Turbine Report No. A-302, Oct. 1946.

3. B. Pinkel and L. R. Turner, "Thermodynamic Data for the Computation of the Performance of Exhaust Gas Turbines," NACA Advance Restricted Report 4B25.

4. Curtiss Wright Corporation, "A Simplified Method of Calculating and Presenting Turbo-Jet Performance," Confidential Report No. M-46-3, Oct. 1946.

5. Glenn L. Martin Co., "Acceleration Characteristics of Westinghouse Model X19B Turbo-Jet Engine," Flight Report No. 37-6233, Jan. 16, 1946.

6. Glenn L. Martin Co., "Report on the Operational Characteristics of the Westinghouse Model X19B Turbo-Jet Engine Installed in a JM-1 Flying Test Bed", Engineering Report No. 2482, Oct. 15, 1946.

RESULTS

PROCEDURE FOR COMPUTATION OF TURBOJET ACCELERATION

1. Given the following atmospheric flight and diffuser conditions:

$$P_0, \; T_0, \; M_0, \; \eta_d$$

2. Assume the independent variables n and $\dot{m}$.*

3. Compute M_2, T_2, and P_2:

$$\rho_2 = \rho_0 \left[\frac{1 + \frac{\gamma-1}{2} M_0^2}{1 + \frac{\gamma-1}{2} M_2^2} \right]^{\left(\frac{\eta_d \gamma}{\gamma-1} - 1\right)} \qquad T_2 = T_0 \left[\frac{1 + \frac{\gamma-1}{2} M_0^2}{1 + \frac{\gamma-1}{2} M_2^2} \right]$$

$$P_2 = P_0 \left[\frac{1 + \frac{\gamma-1}{2} M_0^2}{1 + \frac{\gamma-1}{2} M_2^2} \right]^{\left(\frac{\eta_d \gamma}{\gamma-1}\right)} \qquad M_2 = \frac{\dot{m}}{A_2 \rho_2 \sqrt{\gamma R T_2}}$$

4. Compute non-dimensional compressor parameters Γ_c and σ_c:

$$\Gamma_c = M_2 \sqrt{\frac{\gamma}{2}}$$

$$\sigma_c = \frac{D_c n}{a_2}$$

5. Enter compressor performance charts for the particular turbojet under consideration (See Figures 13, 14, and 15). From these charts determine:

   a) Compressor pressure ratio, $\frac{P_3}{P_2}$

   b) Compressor power coefficient, Ω_c

   c) Compressor temperature ratio, $\frac{T_3}{T_2}$

6. From the above data determine the following:

   a) $P_3 = P_2 \left(\frac{P_3}{P_2} \right)$

   b) $W_c = \Omega_c P_2 A_c a_2$

   c) $T_3 = T_2 \left(\frac{T_3}{T_2} \right)$

*See discussion on selection of $\dot{m}$

7. Enter combustion chamber performance chart for the particular turbojet under consideration and determine pressure ratio P_4/P_3. (See Figure 1... Compute P_4:

$$P_4 = P_3\left(\frac{P_4}{P_3}\right)$$

8. Compute Γ_t and σ_t:

$$\Gamma_t = \frac{\dot{m}}{P_4 A_t}\sqrt{\frac{RT_4}{2}}$$

$$\sigma_t = \frac{D_t n}{a_4}$$

Note that Γ_t involves introducing another independent variable, T_4. For analysis of this subject see discussion.

9. Enter turbine performance charts for the particular turbojet under consideration (See Figure 16, 17, 18 and 19). From these charts determine:

a) $\frac{P_5}{P_4}$

b) Ω_t

c) $\frac{T_5}{T_4}$

d) M_5

10. From these data compute the following:

a) $P_5 = P_4\left(\frac{P_5}{P_4}\right)$

b) $W_t = \Omega_t P_4 A_t a_4$

c) $T_5 = T_4\left(\frac{T_5}{T_4}\right)$

11. Compute nozzle pressure ratio

$$\frac{P_6}{P_5} = \frac{P_o}{P_5}$$

12. Enter nozzle performance chart (See Figure 20) and determine M_6, T_6/T_5 and A_6/A_5. Each installation will have a different chart depending upon nozzle geometry. However, for short tail pipes and

small area ratios flow is nearly isentropic. When nozzle efficiency is known, the foregoing quantities may be determined by computation similar to those for the diffuser.

13. From these data compute the following:

$$T_6 = T_5\left(\frac{T_6}{T_5}\right)$$

$$V_6 = M_6\sqrt{\gamma R T_6}$$

14. Turbojet acceleration

$$\dot{n} = \frac{W_t - W_c}{4\pi^2 n I_p}$$

15. Turbojet thrust

a) When $M_6 < 1.0$:

$$F = \dot{m}\left(V_6 - V_0\right)$$

b) When P_{55}/P_6 exceeds the critical pressure ratio and $M_6 = 1.0$:

$$F = \dot{m}\left(a_6 - V_0\right) + A_6\left(P_6 - P_0\right)$$

(See Appendix II)

Application of this method to the particular cases selected for demonstration produced the ultimate results shown in Figures 1, 2, 3a, and 3b. The effects of tail area ratio and rotor speed on thrust and on acceleration are shown in Figures 1 and 2, which were derived from Figures 7-10. Figures 3a and 3b show effects of rotor speed and tail area ratio on time required to accelerate to a particular engine speed.

DISCUSSION

The method of computation proposed in this paper is simple in form, but the calculations are lengthy and the method requires some practice to estimate quickly the proper range of mass flows to select for the calculations. This is to be expected since the mass flow is not, in fact, an independent variable. However it has been chosen as such for the purpose of ease of computation. The independent variables in the unsteady state condition are: (1) Flight conditions (velocity, density, temperature and pressure) (2) Rotor speed (3) Fuel mass rate of flow and (4) Tail area ratio. (This contrasts with the steady state or equilibrium condition where the rotor speed is a dependent variable).

In the method outlined in this paper the fuel mass rate of flow has been replaced as an independent variable by the use of a constant turbine inlet temperature. The independent variables used then are: (1) Flight conditions (2) Air mass rate of flow (3) Turbine inlet temperature and (4) Rotor speed. Since the flight conditions have been held constant throughout the series of computations made here, there remain only three independent variables.

As a starting point in the calculation a rotor speed and turbine inlet temperature are selected. Then for each air mass flow chosen, values of tail area ratio, acceleration, and thrust are obtained. This makes tail area ratio, acceleration, and thrust a function of the air mass rate of flow. Other sets of calculations may be obtained by varying the rotor speed and the turbine inlet temperature and repeating the procedure.

The geometric configuration of the unit imposes certain natural limitations upon the selection of the air mass rate of flow. If $\dot{m}$ is chosen too low, the compressor operates in a stalled conditions. This is obviously undesirable. Under certain conditions of higher mass flow a Mach number of unity is reached at some point in the unit and a condition of critical flow exists due to sonic velocity in the turbine nozzle throat. This occurs when the value of $\sqrt{\tau}$ exceeds a critical value (In this case 0.482) and is clearly shown in Figure 10. Under certain conditions of high $\dot{m}$ when the critical flow limit is not reached, a value of $\dot{m}$ may be chosen so high that the stagnation pressure in the tail pipe is less than the atmospheric pressure. This obviously is a physically impossible condition and occurs when too high a value of $\dot{m}$ is selected. The fallacy does not become apparent in the calculations until the point of entry into the nozzle chart (Figure 20), when the tail area ratio appears to be something "greater than infinite".

The turbine inlet temperature is controlled directly by the mass rate of fuel flow into the combustion chamber. However, since the calculation of the turbine inlet temperature is a process involving the combustion efficiencies and the heating value of the fuel it was not considered to be within the scope of this report to carry out these calculations. The assumption was therefore made that a sufficient amount of fuel was consumed in order to provide the required T_4. In order to ascertain the effects of turbine inlet temperature variation, calculations were for a T_4 of 2000°R which is approximately the maximum

allowable for continuous operation of the Westinghouse X19B ($1960^{o}R$) and for a T_4 of $200^{o}F$ higher.

Exact information regarding pressure drop across the combustion chamber was not available until after calculations were completed. The estimate used in these calculations ($\Delta P/P_3 = 0.02$) was subsequently found to agree reasonably well with data from tests made on the combustion chamber of this unit by Westinghouse. (See Figure 12).

No allowance has been made in this analysis for mass of fuel added, air leakage between compressor and turbine, or power required for accessories. The effects of these small quantities tend to compensate each other.

Figures 1, and 2 show the effects on acceleration and thrust of variation of rotor speed and tail area ratio at a particular turbine inlet temperature. Comparison of these two charts shows that temporarily exceeding the peak continuous allowable temperatures of the Westinghouse X19B produces a slight increase in acceleration at low rotor speed, but also introduces the danger of operating within the compressor stall.

It is interesting to note that acceleration may be obtained only at the expense of thrust. This further aggravates the problem of thrust requirements under emergency conditions.

Convergence of lines of constant tail area ratio as their magnitude increases shows that increasing tail area ratio above 2.0 produces only slight gain in acceleration.

Figures 3a and 3b show the effects of rotor speed and tail area ratio on time required to accelerate to a particular engine speed. Because the accelerations is a function of the small difference between

large turbine and compressor powers, accuracy of the results is subject to question, and should be verified by experimental data.

Pertinent extracts from a report on flight tests of a X19B unit conducted by the Glenn L. Martin Company are shown in Figure 6. Since manual control of fuel flow was used in these tests, a constant turbine inlet temperature could not be maintained. An effort was made, however, to follow a procedure which would produce maximum acceleration. No rational estimate of turbine inlet temperatures used in the tests is possible. However, since maximum acceleration was their goal, it is presumed that these temperatures were at or near the maximum allowable (1960°R) during most of the run. Tail area ratio was held constant, but at a value not specified. However, estimates derived from a photograph of this installation indicate the tail area ratio was approximately 2.0. Acceleration times, then, would be roughly comparable to those shown in Figure 3a, for a tail area ratio of approximately 2.0. The agreement is close enough that quantitative values may be considered roughly correct, and the qualitative effects may be considered reliable.

CONCLUSIONS

1. Acceleration characteristics of a turbojet may be computed by the method presented in this report, provided adequate experimental data for component parts is available.

2. Results obtained through use of this method agree closely enough with experimental data so that quantitative values obtained may be considered roughly correct, and qualitative effects may be considered reliable.

3. This method is not applicable for prediction of a schedule of acceleration for a turbojet under actual operating conditions until extensive calculations have been made over a complete range of turbine inlet temperature and flight conditions.

4. Increasing tail area ratio increases acceleration but with diminishing effect as tail area ratio gets larger.

5. Increasing peak temperature increases acceleration at all rotor speeds. The relative increase is much greater at higher speeds.

6. Increasing peak temperature tends to induce earlier compressor stall.

7. Acceleration of a turbojet under any condition is slow when compared to a reciprocating engine.

REFERENCES

1. "Jet Propulsion", A reference text prepared by the staffs of the Guggenheim Aeronautical Laboratory and the Jet Propulsion Laboratory, GALCIT, California Institute of Technology, for the Air Technical Service Command. (Restricted)

2. Westinghouse Aviation Gas Turbine Report No. A-302, Oct. 1946.

3. B. Pinkel and L. R. Turner, "Thermodynamic Data for the Computation of the Performance of Exhaust Gas Turbines," NACA Advance Restricted Report 4B25.

4. Curtiss Wright Corporation, "A Simplified Method of Calculating and Presenting Turbo-Jet Performance," Confidential Report No. M-46-3, Oct. 1946.

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6. Glenn L. Martin Co., "Report on the Operational Characteristics of the Westinghouse Model X19B Turbo-Jet Engine Installed in a JM-1 Flying Test Bed", Engineering Report No. 2482, Oct. 15, 1946.

FIGURE 1

TURBOJET ROTOR ACCELERATION VS. TAIL AREA RATIO AT VARIOUS ROTOR SPEEDS

TURBINE INLET TEMPERATURE - 2000°R
SEA LEVEL STANDARD CONDITIONS
FLIGHT VELOCITY - 100 MPH

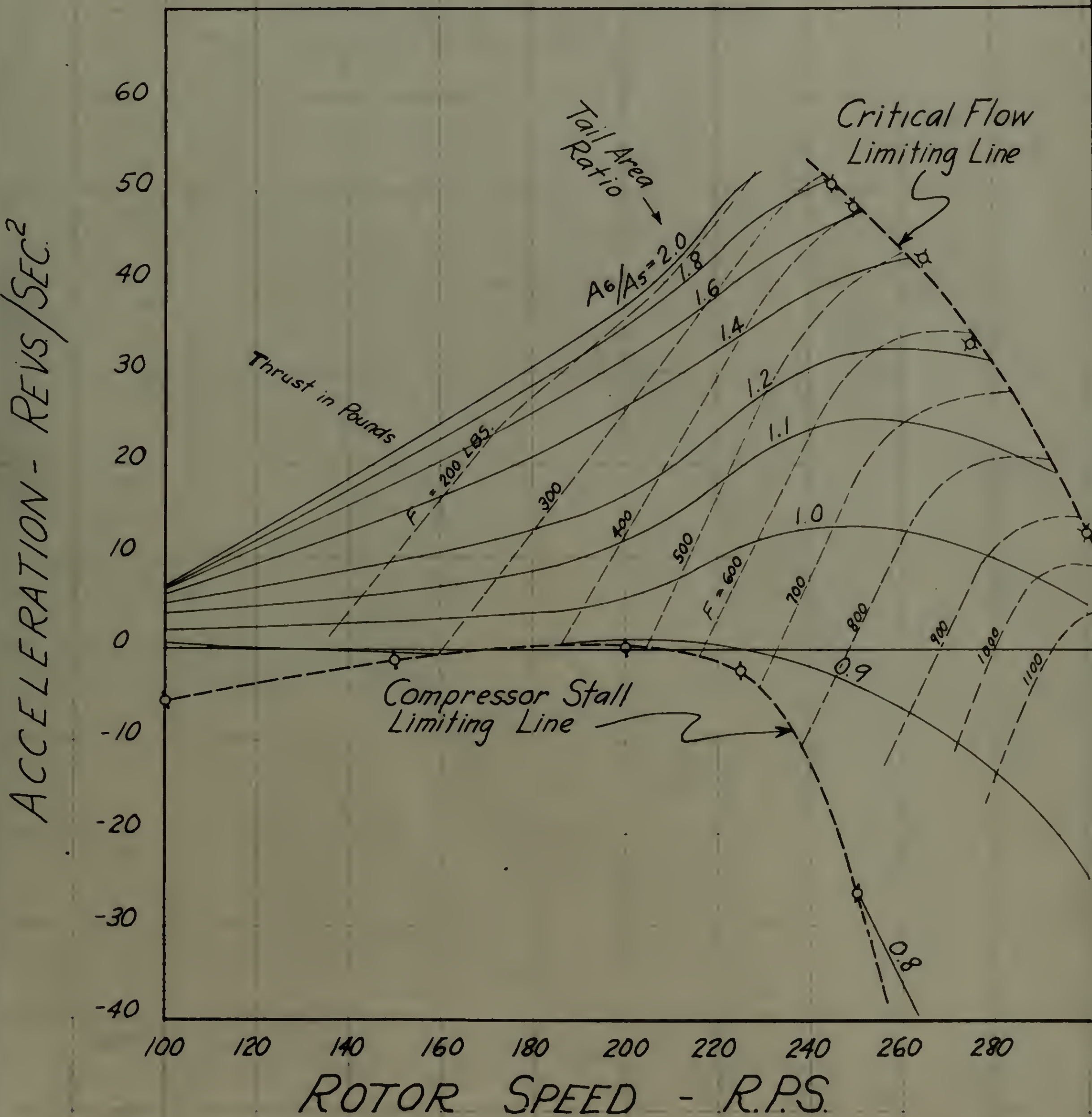


FIGURE 2

TURBOJET ROTOR ACCELERATION VS ROTOR SPEED AT VARIOUS TAIL AREA RATIOS

TURBINE INLET TEMPERATURE 2200°R.
SEA LEVEL STANDARD CONDITIONS
FLIGHT VELOCITY - 100 MPH

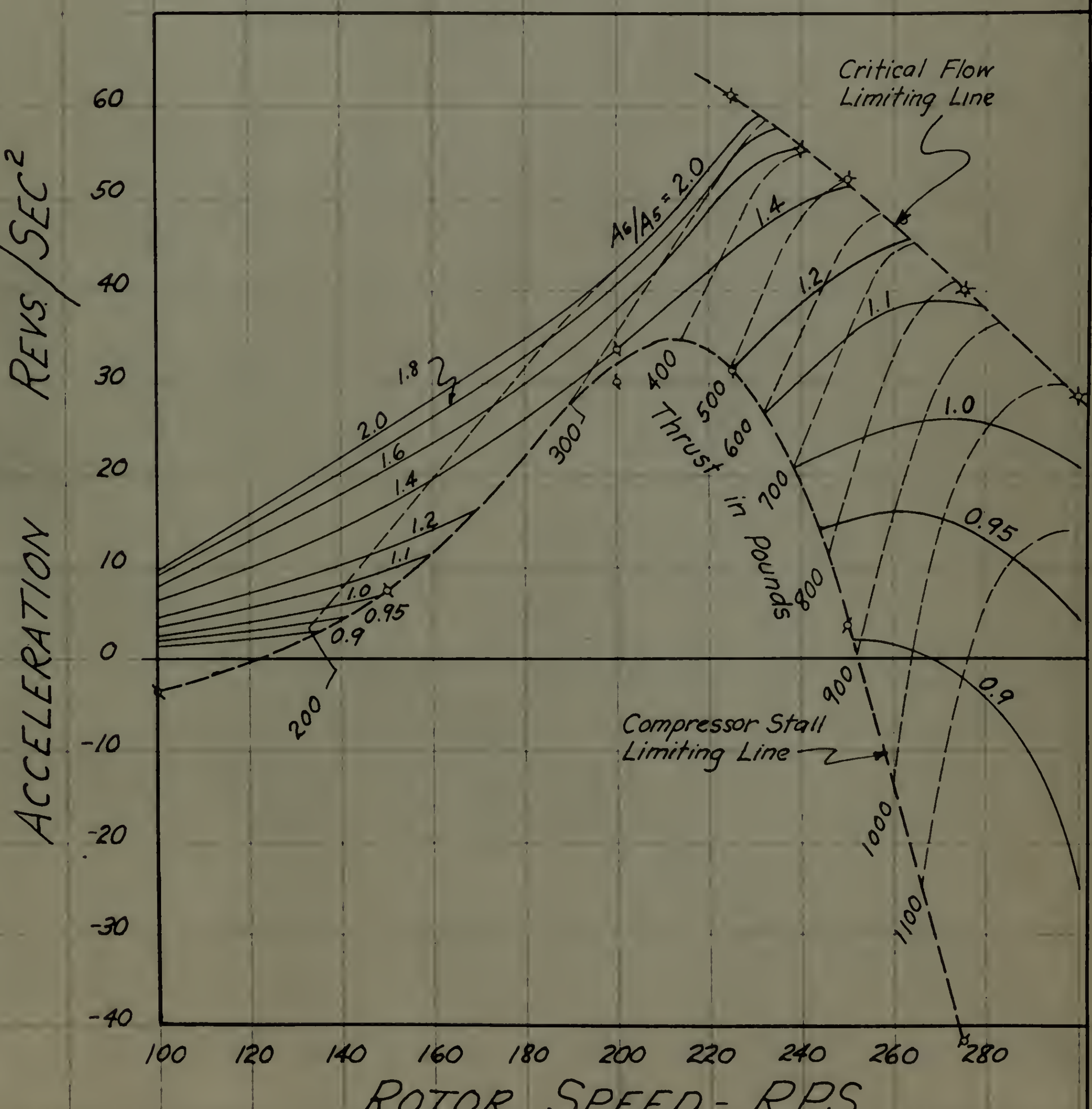


FIGURE 8a

TIME TO ACCELERATE

TURBINE INLET TEMPERATURE - 2000°R
SEA LEVEL STANDARD CONDITIONS
FLIGHT VELOCITY - 100 MPH

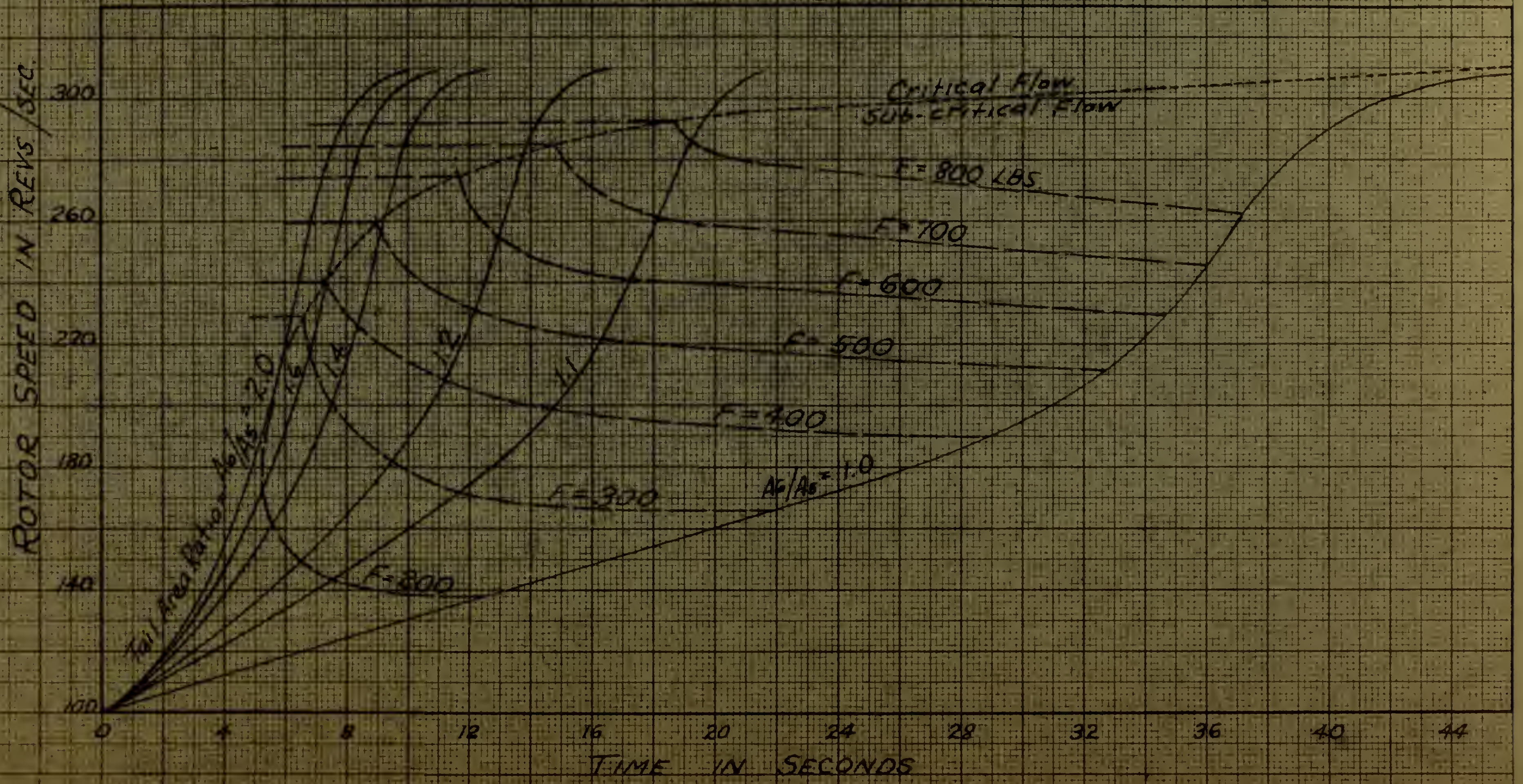


FIGURE 3b

TIME TO ACCELERATE AT VARIOUS TAIL AREA RATIOS

TURBINE INLET TEMPERATURE 2200°R
SEA LEVEL STANDARD CONDITIONS
FLIGHT VELOCITY - 100 MPH

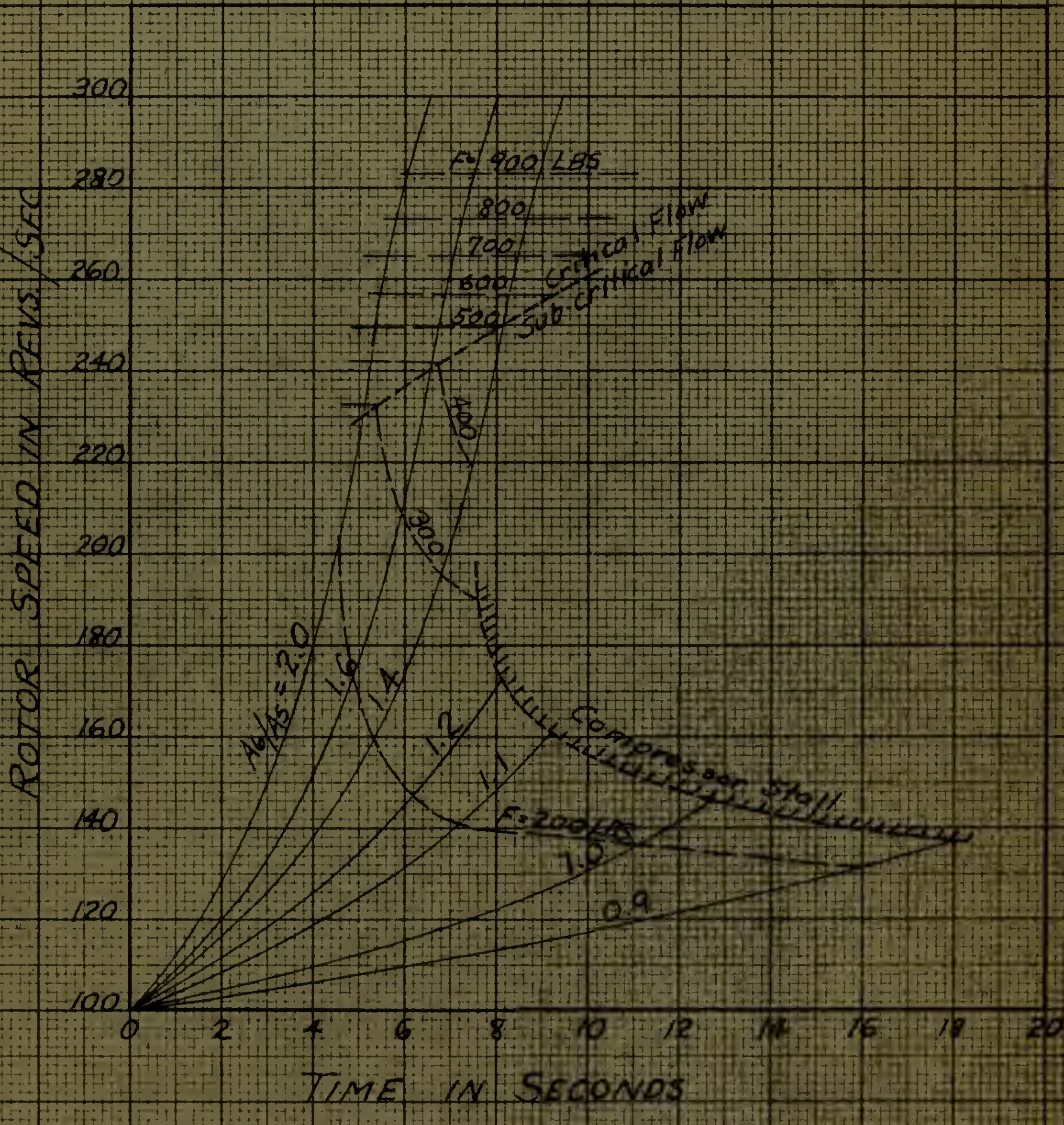


FIGURE 4

TURBOJET ROTOR ACCELERATION VS. TAIL AREA RATIO FOR VARIOUS ROTOR SPEEDS

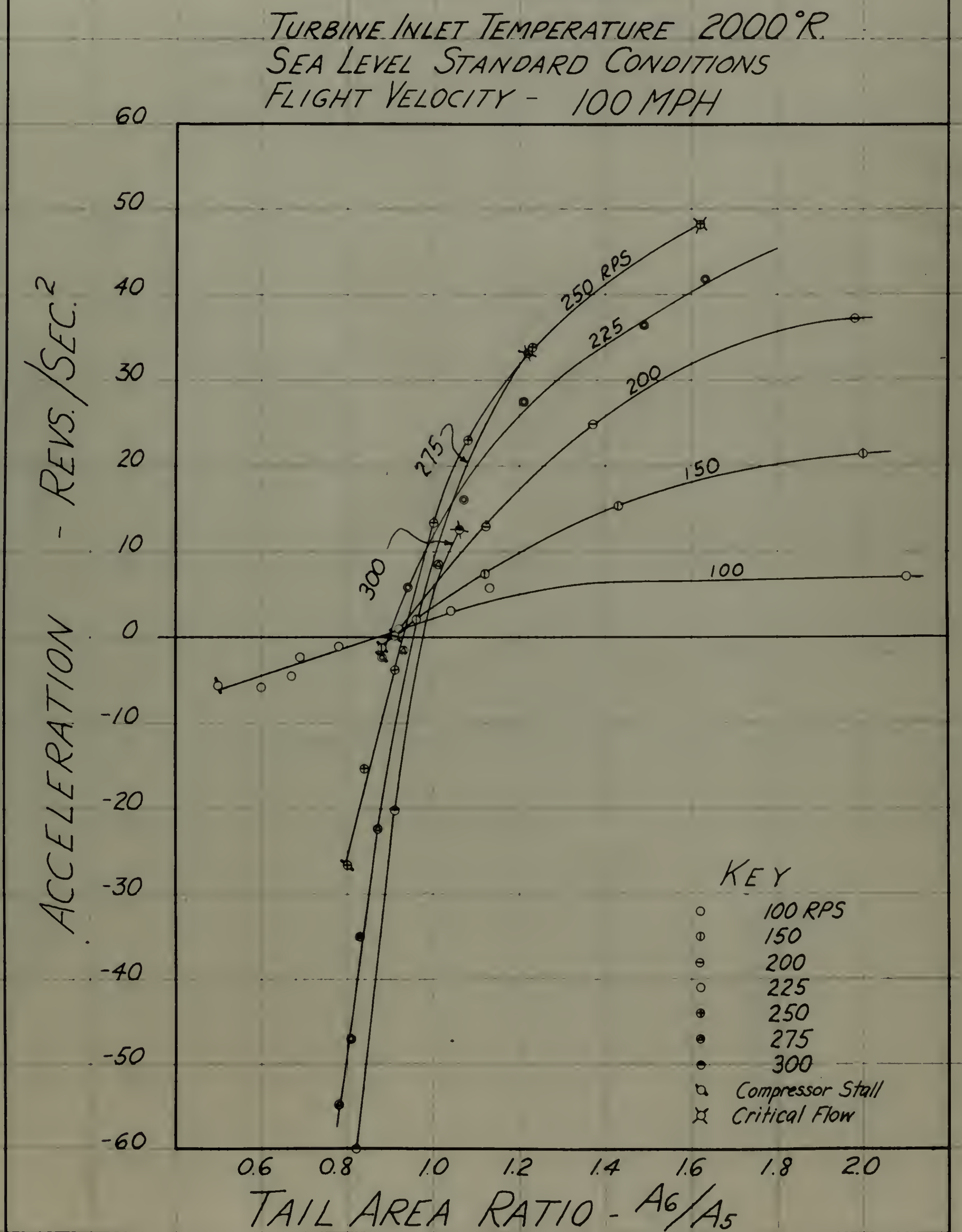


FIGURE 5

TURBOJET ROTOR ACCELERATION VS. TAIL AREA RATIO FOR VARIOUS ROTOR SPEEDS

TURBINE INLET TEMPERATURE 2200°R
SEA LEVEL STANDARD CONDITIONS
FLIGHT VELOCITY - 100 MPH

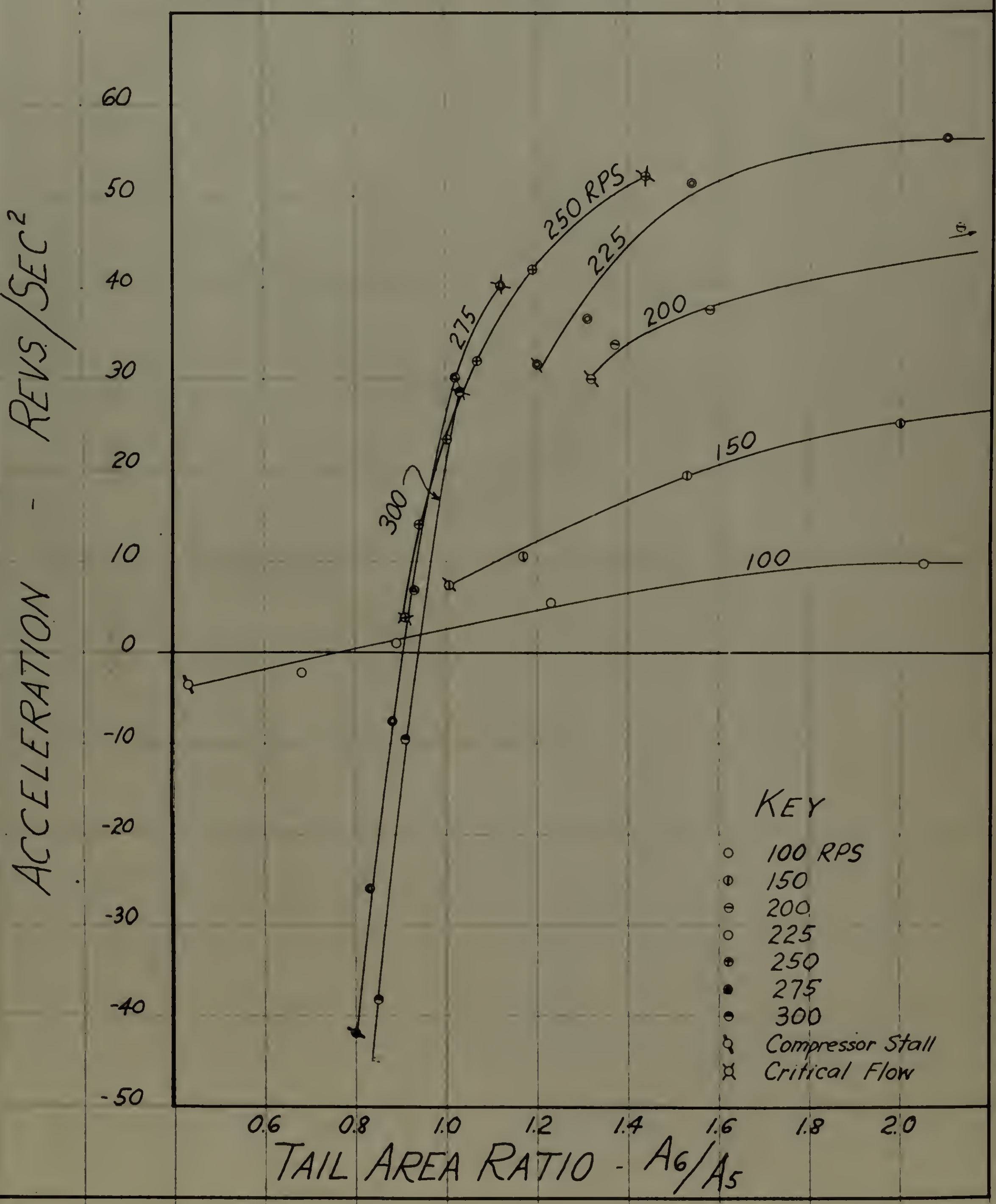


FIGURE 6

EXPERIMENTAL ACCELERATION CURVES

WESTINGHOUSE X19B TURBOJET

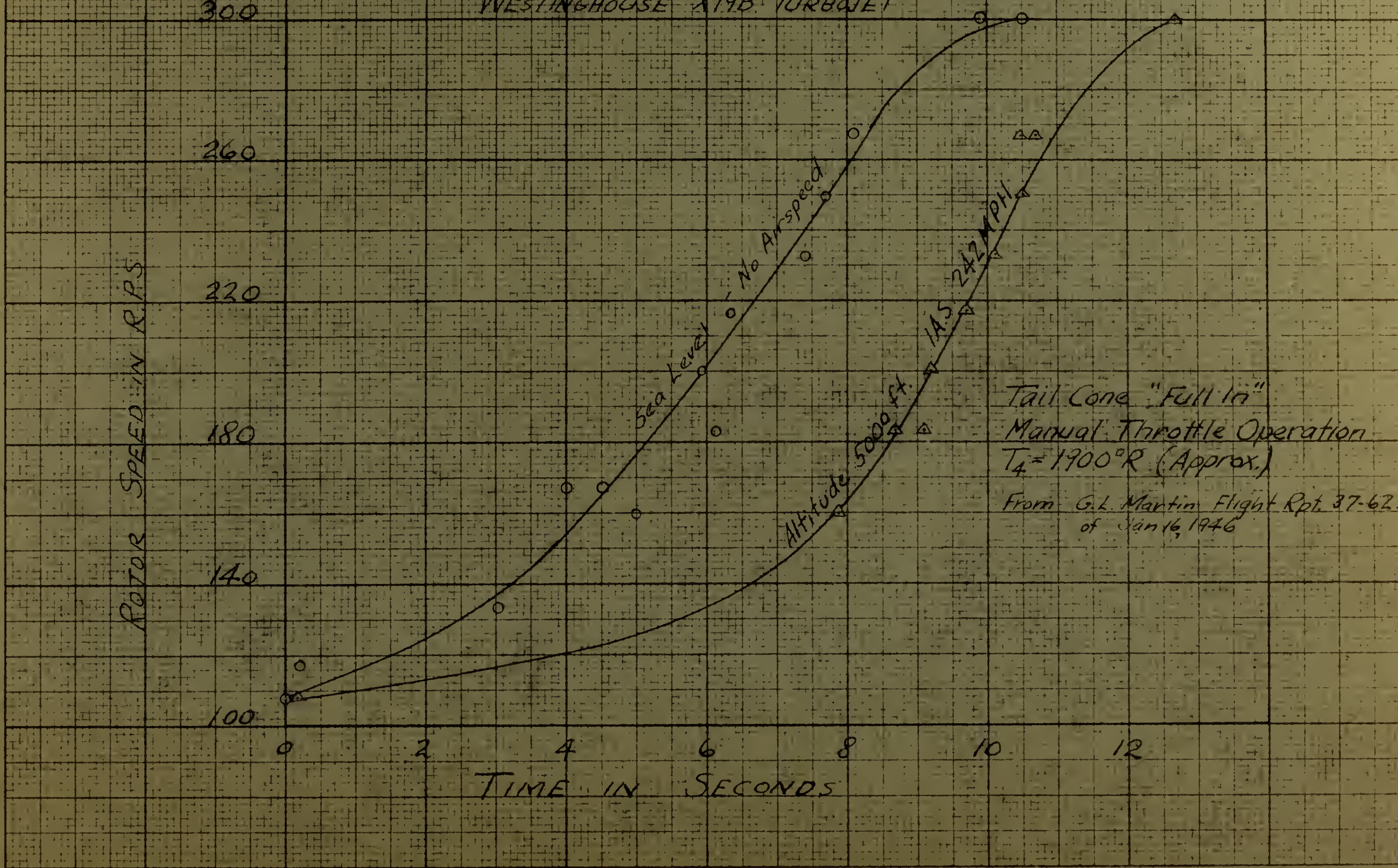


FIGURE 7

TURBOJET THRUST VS TAIL AREA RATIO
FOR VARIOUS ROTOR SPEEDS

TURBINE INLET TEMPERATURE - 2000 °R
SEA LEVEL STANDARD CONDITIONS
FLIGHT VELOCITY - 100 MPH

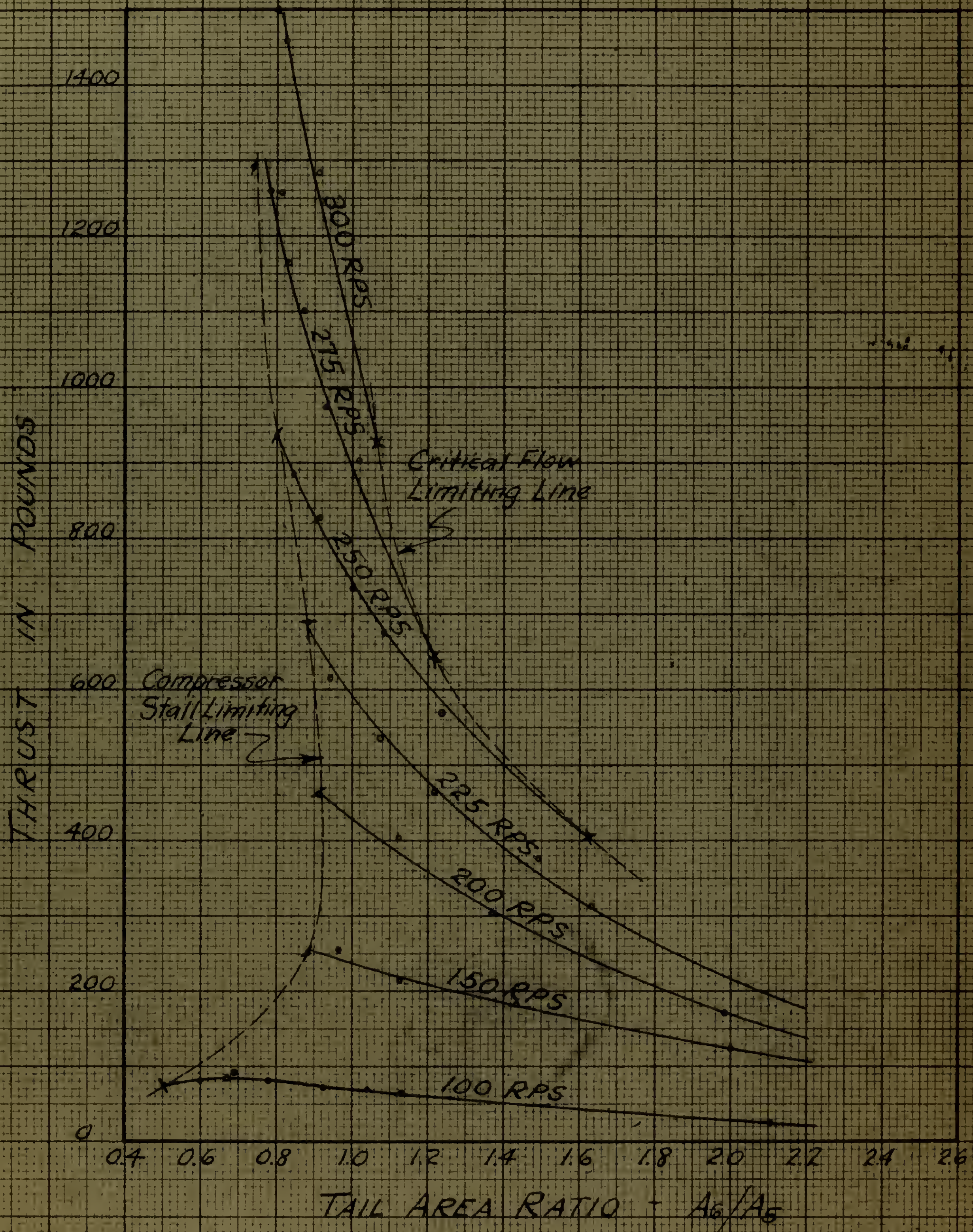


FIGURE 8

TURBOJET THRUST VS ROTOR SPEED FOR VARIOUS TAIL AREA RATIOS

TURBINE INLET TEMPERATURE 2000°R

SEA LEVEL STANDARD CONDITIONS

FLIGHT VELOCITY - 100 MPH

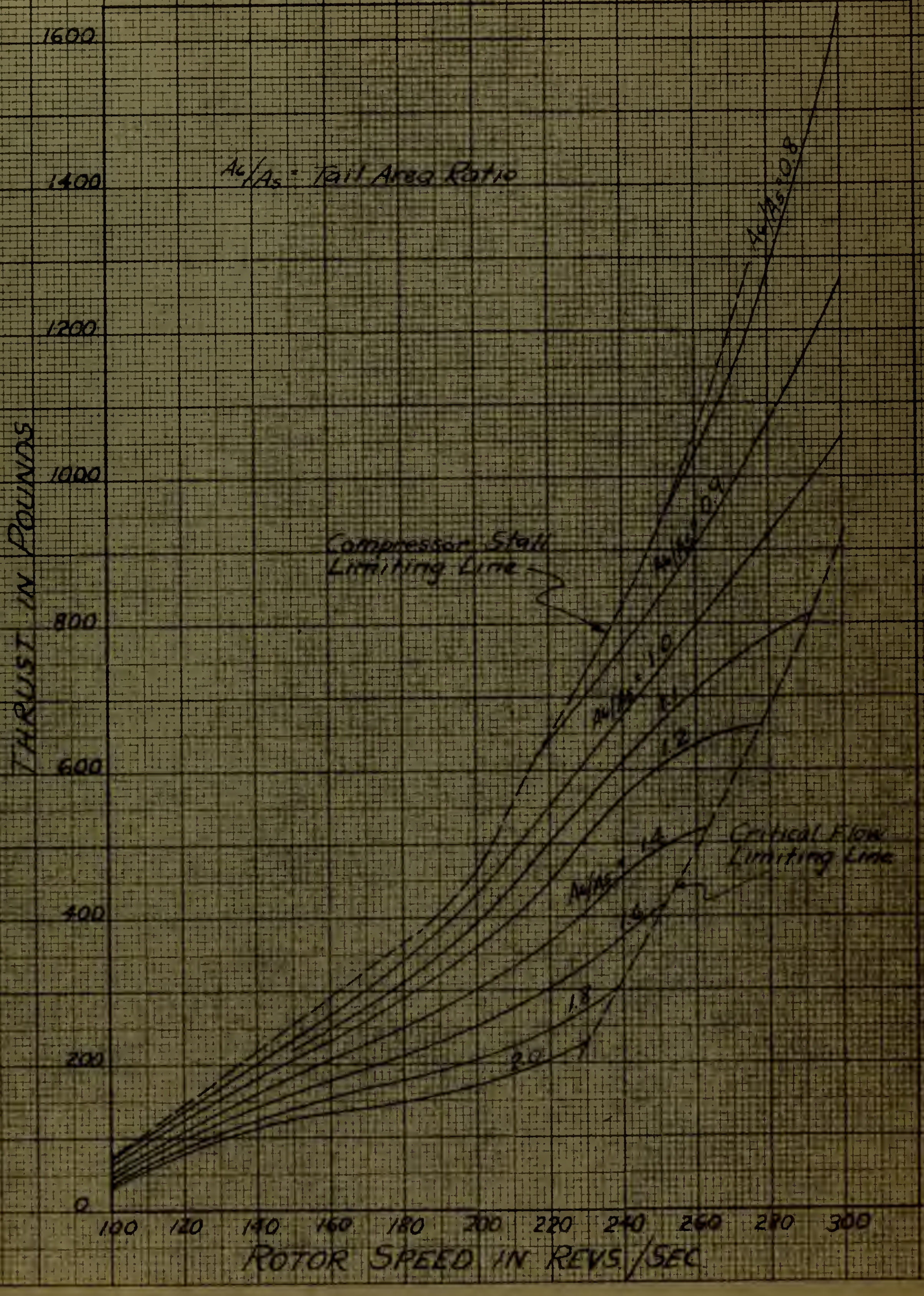


FIGURE 9

TURBOJET THRUST VS TAIL AREA RATIO FOR VARIOUS ROTOR SPEEDS

TURBINE INLET TEMPERATURE - 2200°R
SEA LEVEL STANDARD CONDITIONS
FLIGHT VELOCITY - 100 MPH

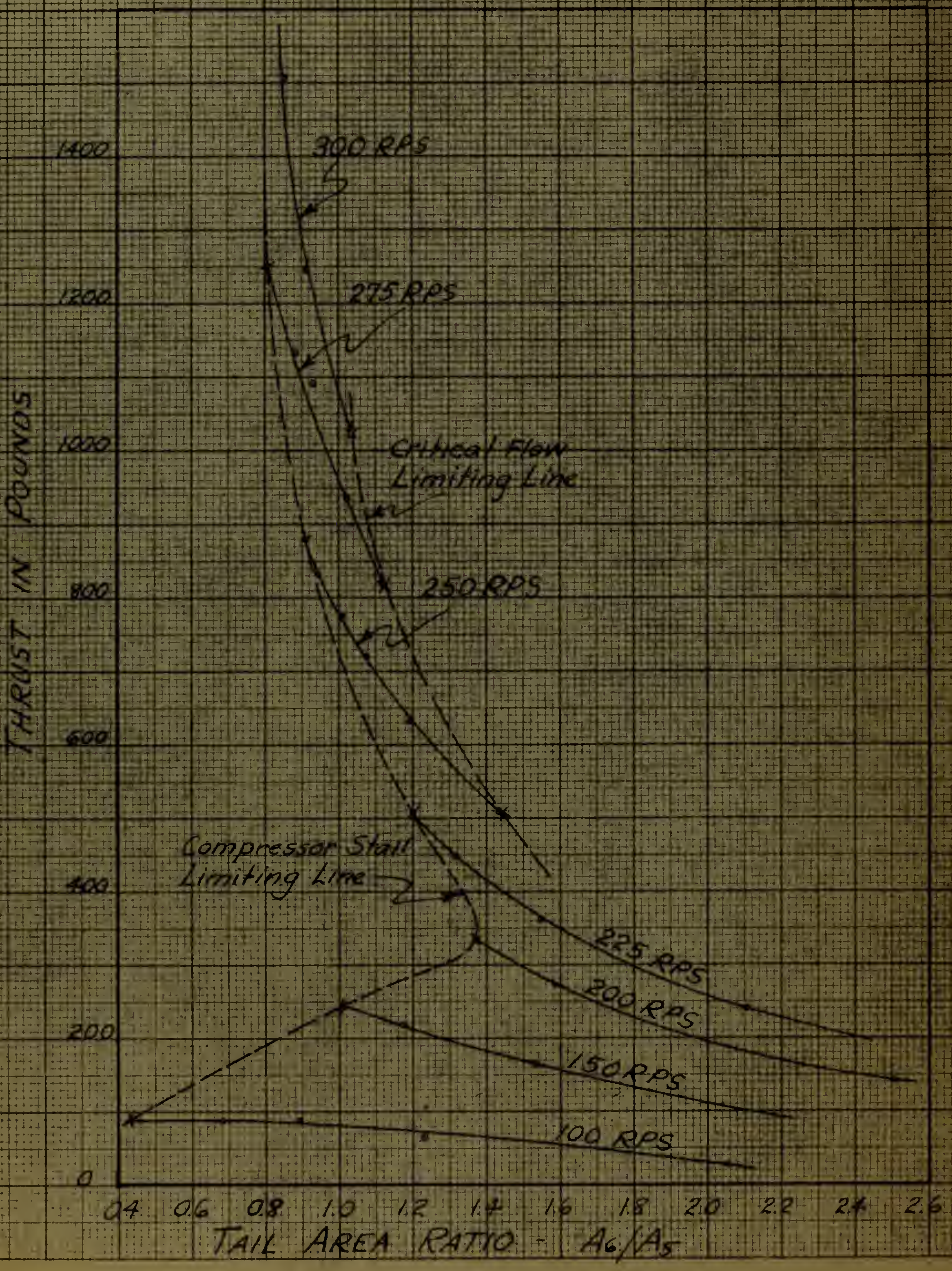


FIGURE 10

TURBOJET THRUST VS ROTOR SPEED FOR VARIOUS TAIL AREA RATIOS

TURBINE INLET TEMPERATURE - 2200° R

SEA LEVEL STANDARD CONDITIONS

FLIGHT VELOCITY - 100 MPH

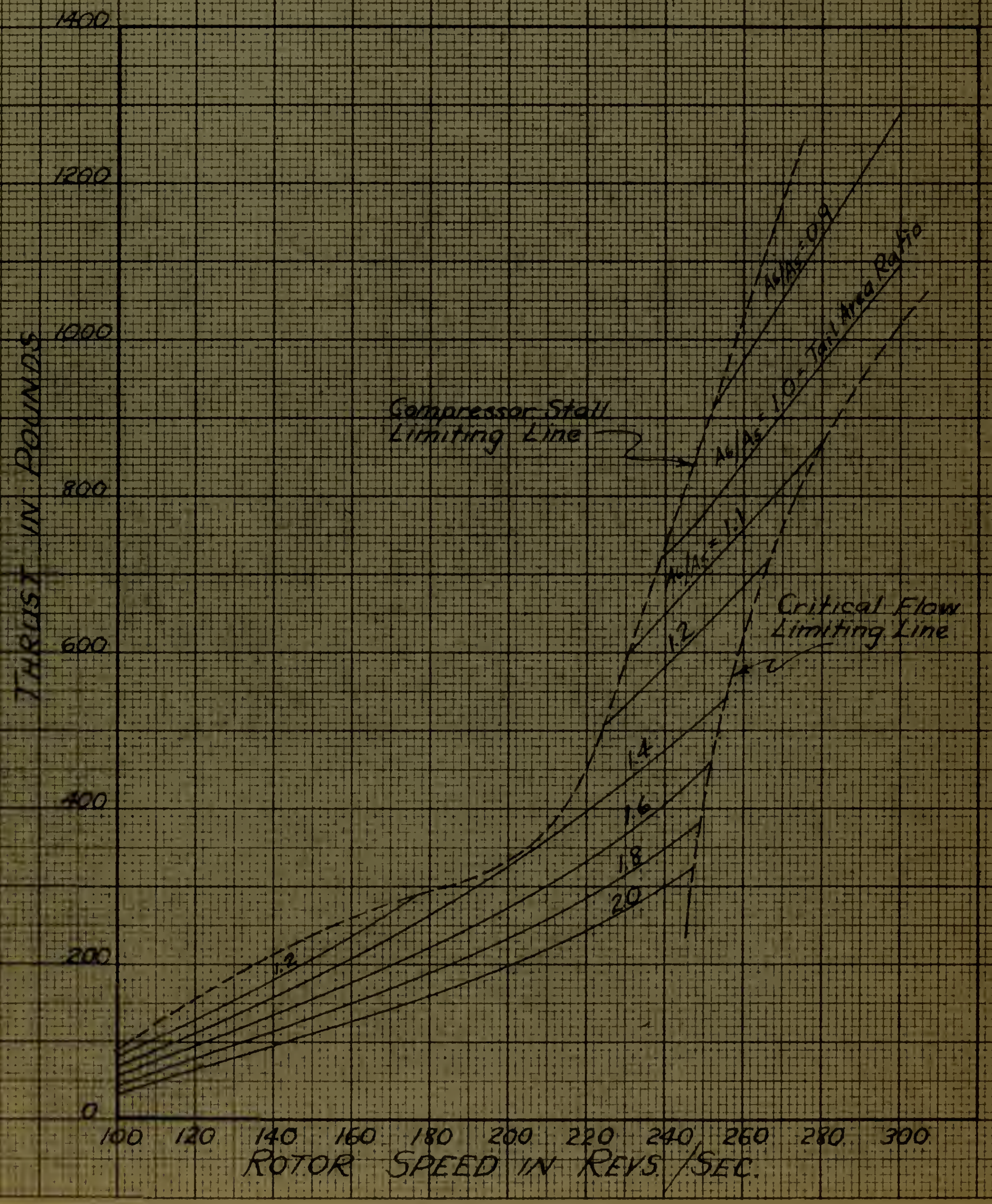


FIGURE 11

COMPRESSOR INLET CONDITIONS

M_2, P_2, T_2 VS MASS FLOW

$M_0 = 0.133$ (100MPH)

SEA LEVEL STANDARD CONDITIONS

ISENTROPIC DIFFUSION

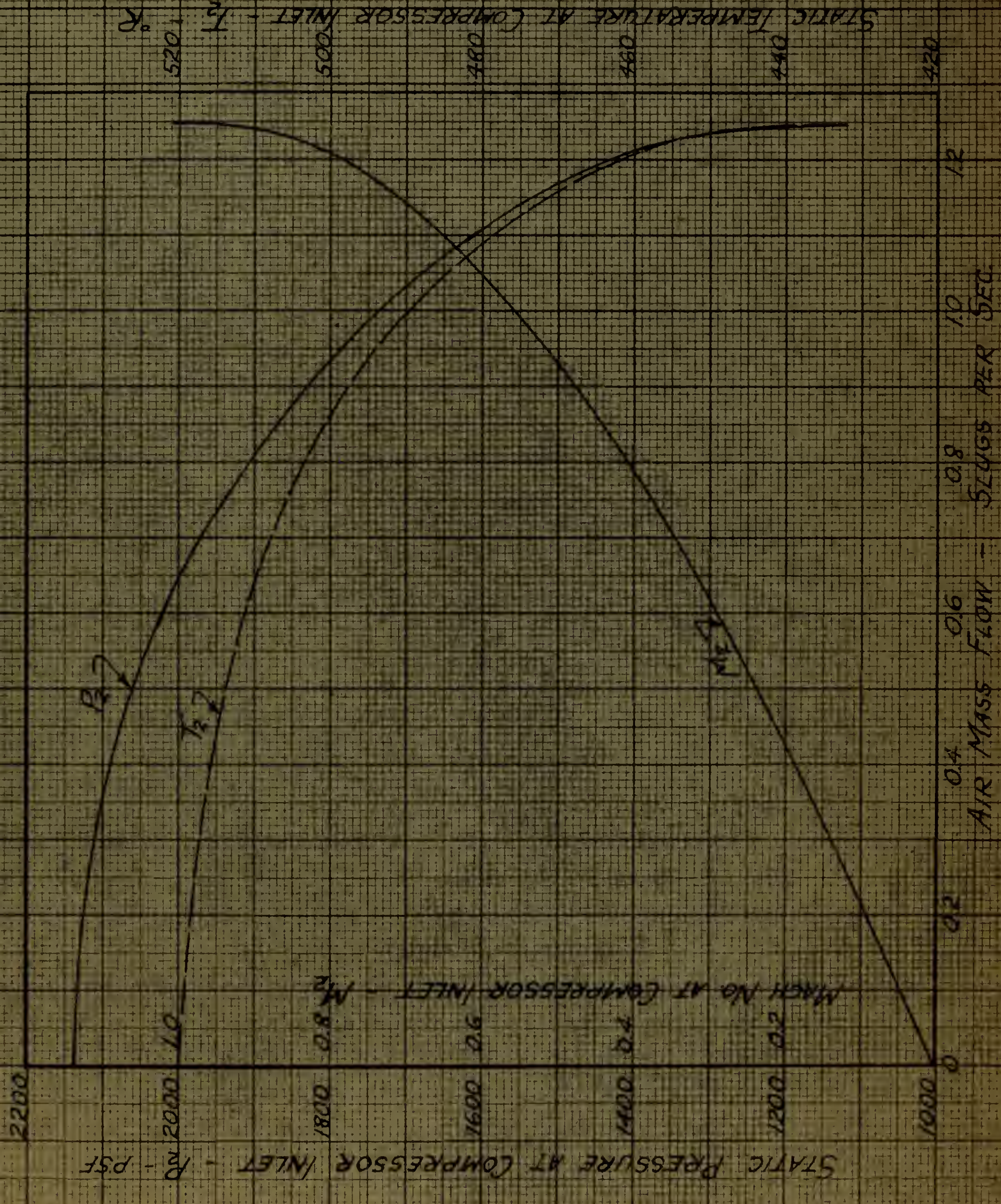


FIGURE 12

TOTAL PRESSURE DROP THROUGH COMBUSTOR

WESTINGHOUSE JET PROPULSION ENGINE

MODEL X-19XB-3

(FROM WESTINGHOUSE AGT RPT. NO. A-302 OF OCT. 46)

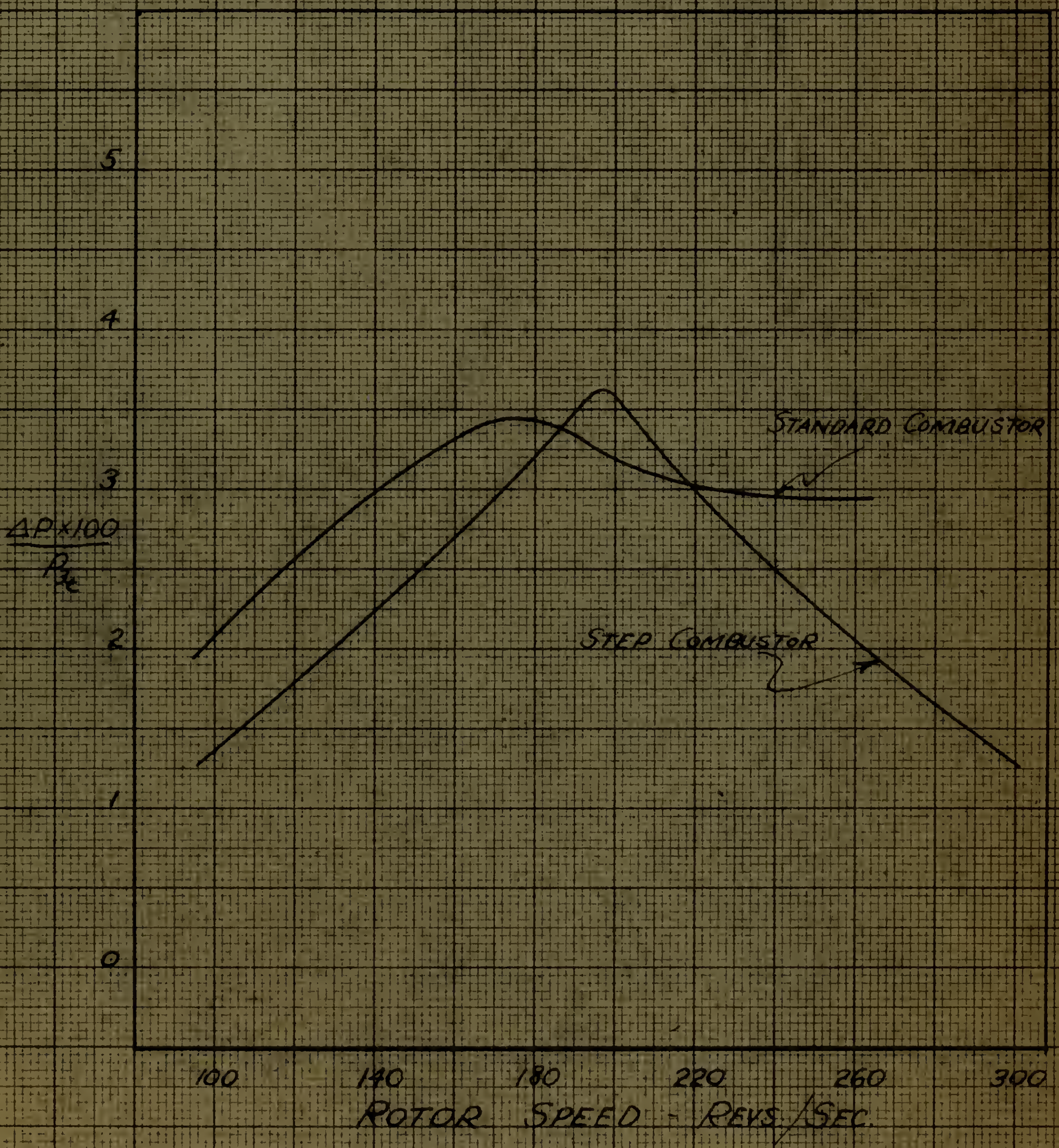
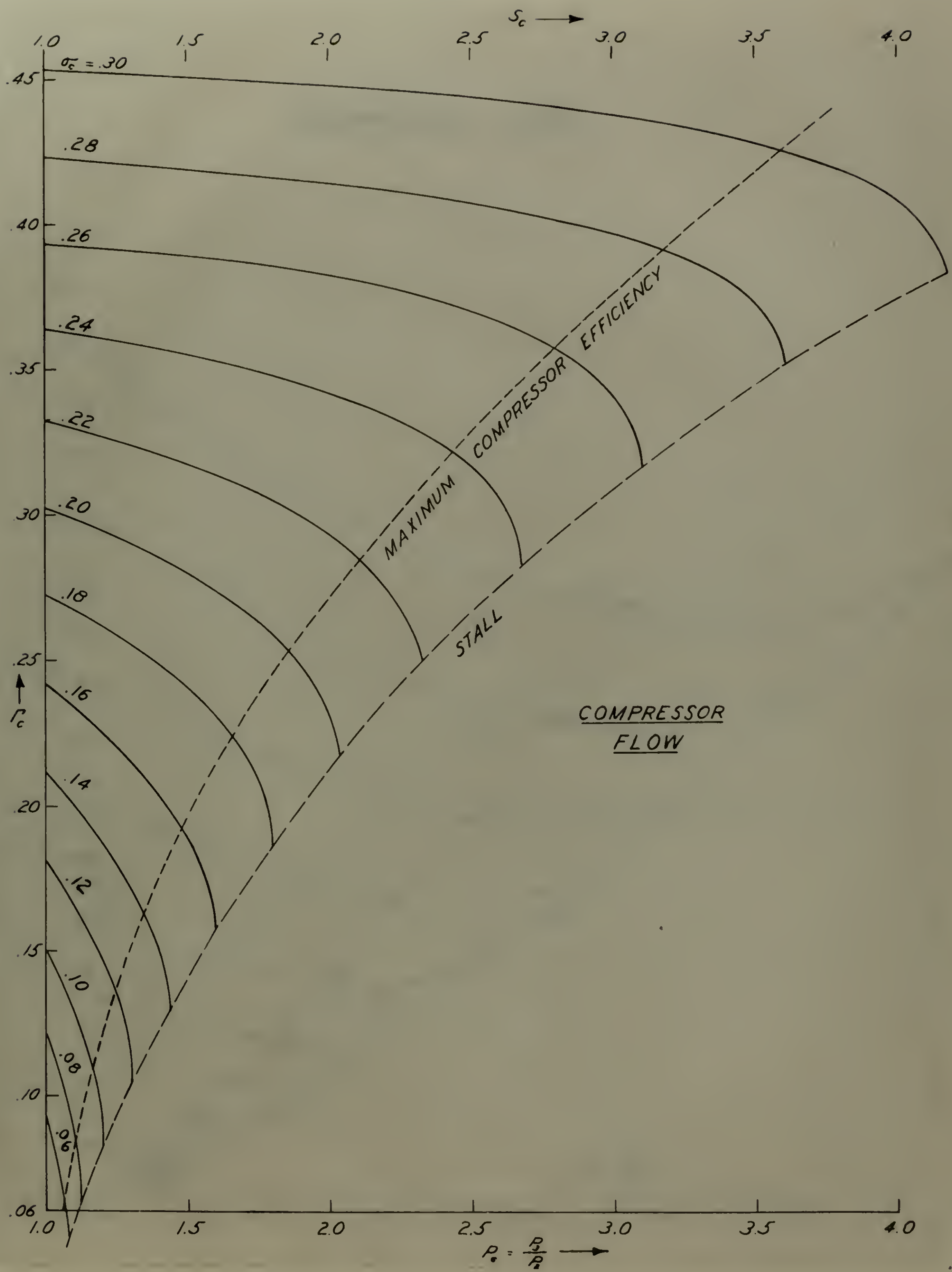


FIGURE 13



S_c
1.0
1.5
2.0
2.5
3.0
3.5
4.0
$\sigma_c = .30$
.28
.26
.24
.22
.20
.18
.16
.14
.12
.10
.08
.06
.45
.40
.35
.30
.25
Γ_c
.20
.15
.10
.06
MAXIMUM COMPRESSOR EFFICIENCY
STALL
COMPRESSOR FLOW
1.0
1.5
2.0
2.5
3.0
3.5
4.0
$P_c = \frac{P_3}{P_2}$

FIGURE 14

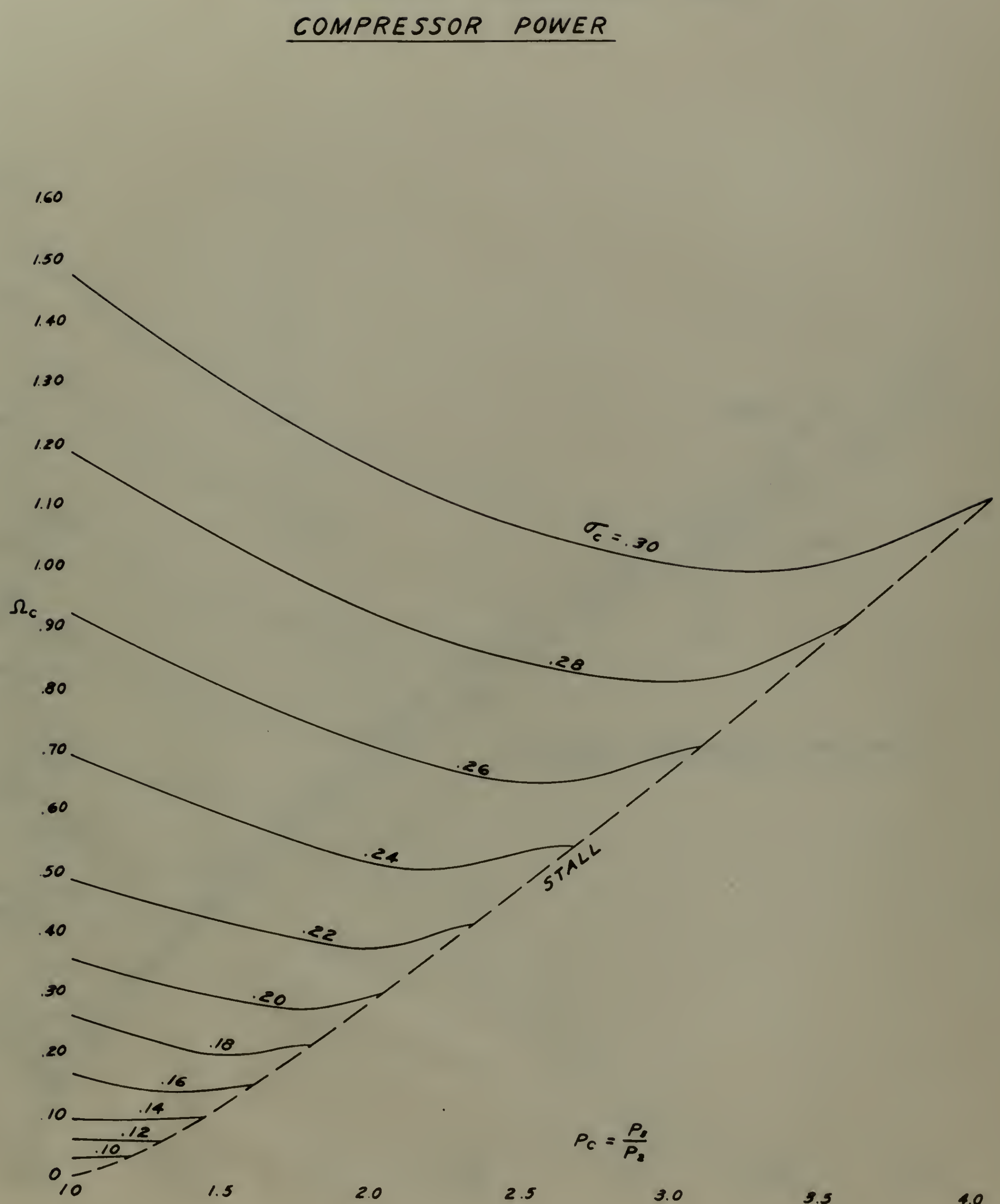
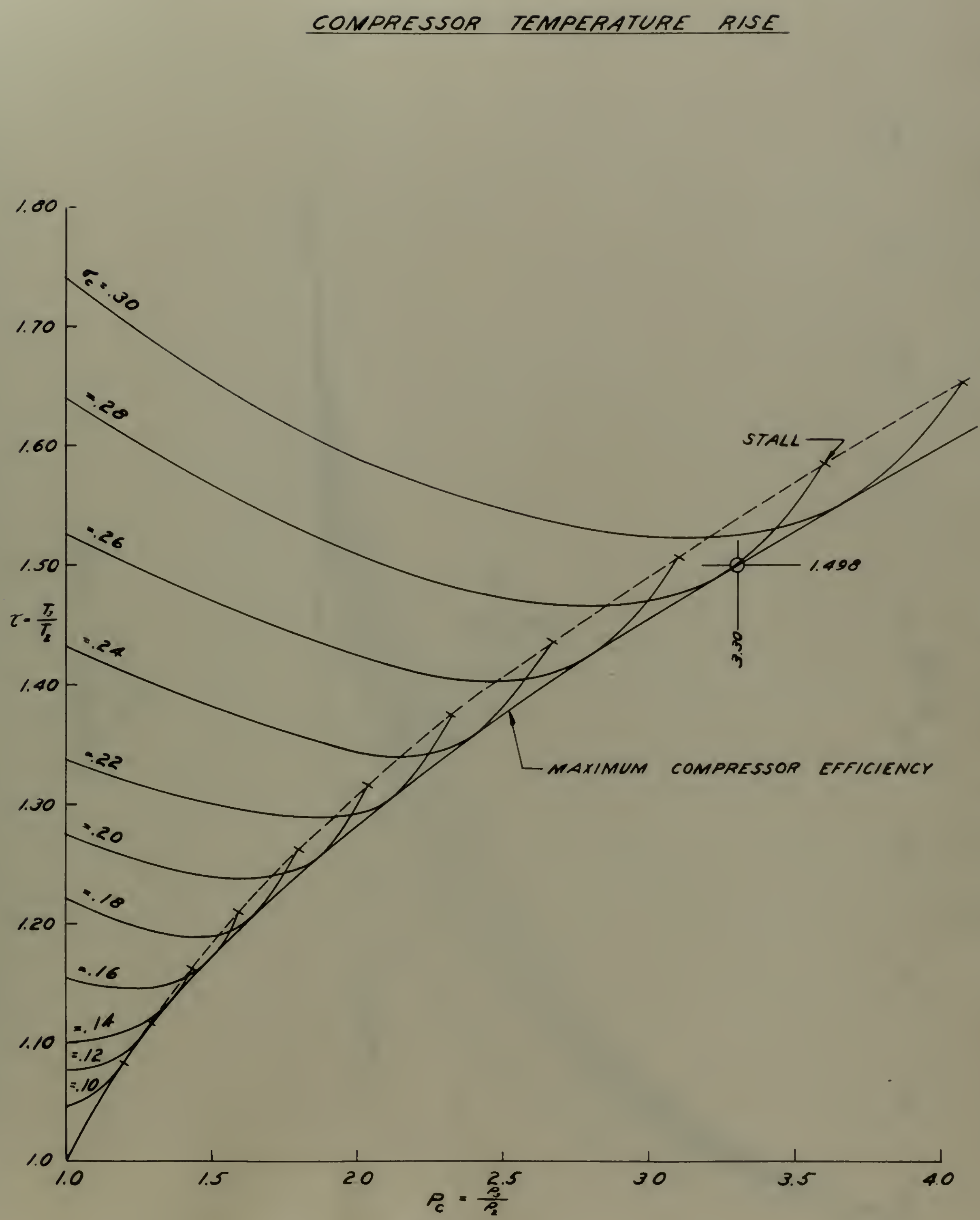
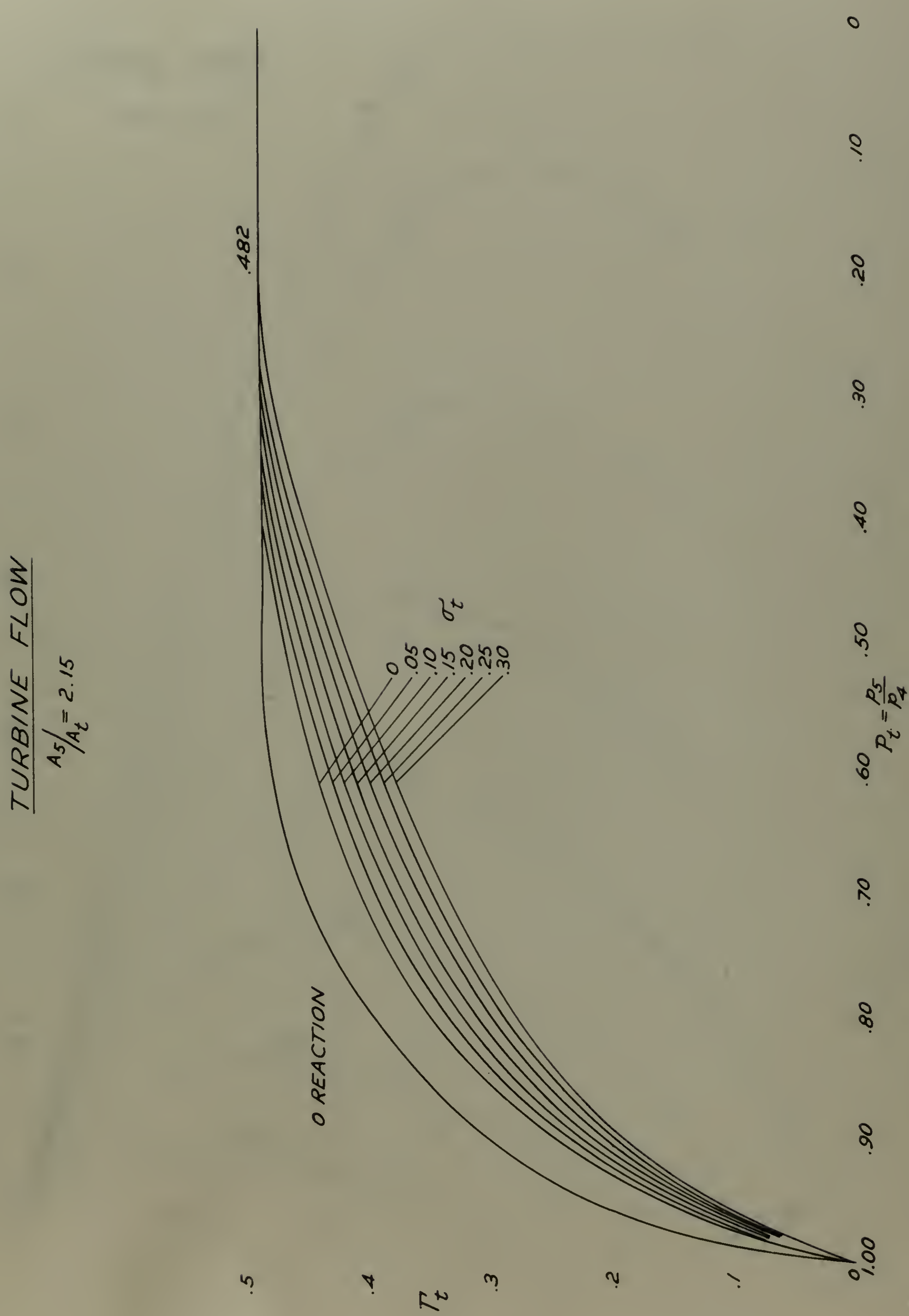

COMPRESSOR POWER
1.60
1.50
1.40
1.30
1.20
1.10
1.00
Ω_c
.90
.80
.70
.60
.50
.40
.30
.20
.10
0
$\sigma_c = .30$
.28
.26
.24
.22
.20
.18
.16
.14
.12
.10
STALL
$P_c = \frac{P_1}{P_2}$
1.0
1.5
2.0
2.5
3.0
3.5
4.0

FIGURE 15



COMPRESSOR TEMPERATURE RISE
1.80
1.70
1.60
1.50
1.40
1.30
1.20
1.10
1.0
$\tau = \frac{T_3}{T_2}$
σ_c = .30
=.28
=.26
=.24
=.22
=.20
=.18
=.16
=.14
=.12
=.10
STALL
1.498
3.30
MAXIMUM COMPRESSOR EFFICIENCY
1.0
1.5
2.0
2.5
3.0
3.5
4.0
$P_c = \frac{P_3}{P_2}$

FIGURE 16



TURBINE FLOW
A5/At = 2.15
0 REACTION
.482
σt
0
.05
.10
.15
.20
.25
.30
Tt
0
.1
.2
.3
.4
.5
1.00
.90
.80
.70
.60
.50
.40
.30
.20
.10
0
Pt = P5/P4

FIGURE 17

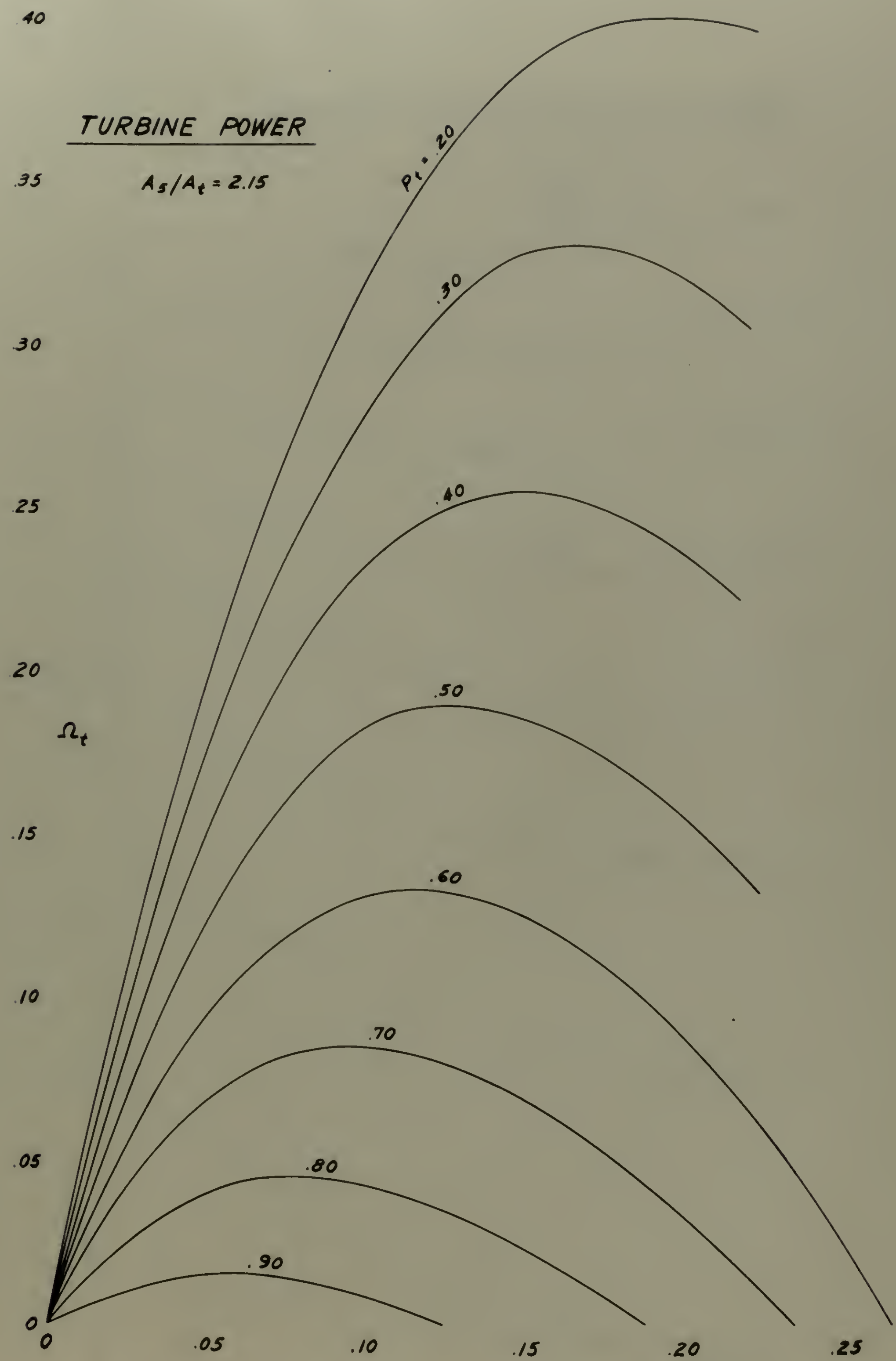

TURBINE POWER
$A_s/A_t = 2.15$
$p_t = .20$
.30
.40
.50
.60
.70
.80
.90
Ω_t
.40
.35
.30
.25
.20
.15
.10
.05
0
0
.05
.10
.15
.20
.25

FIGURE 18

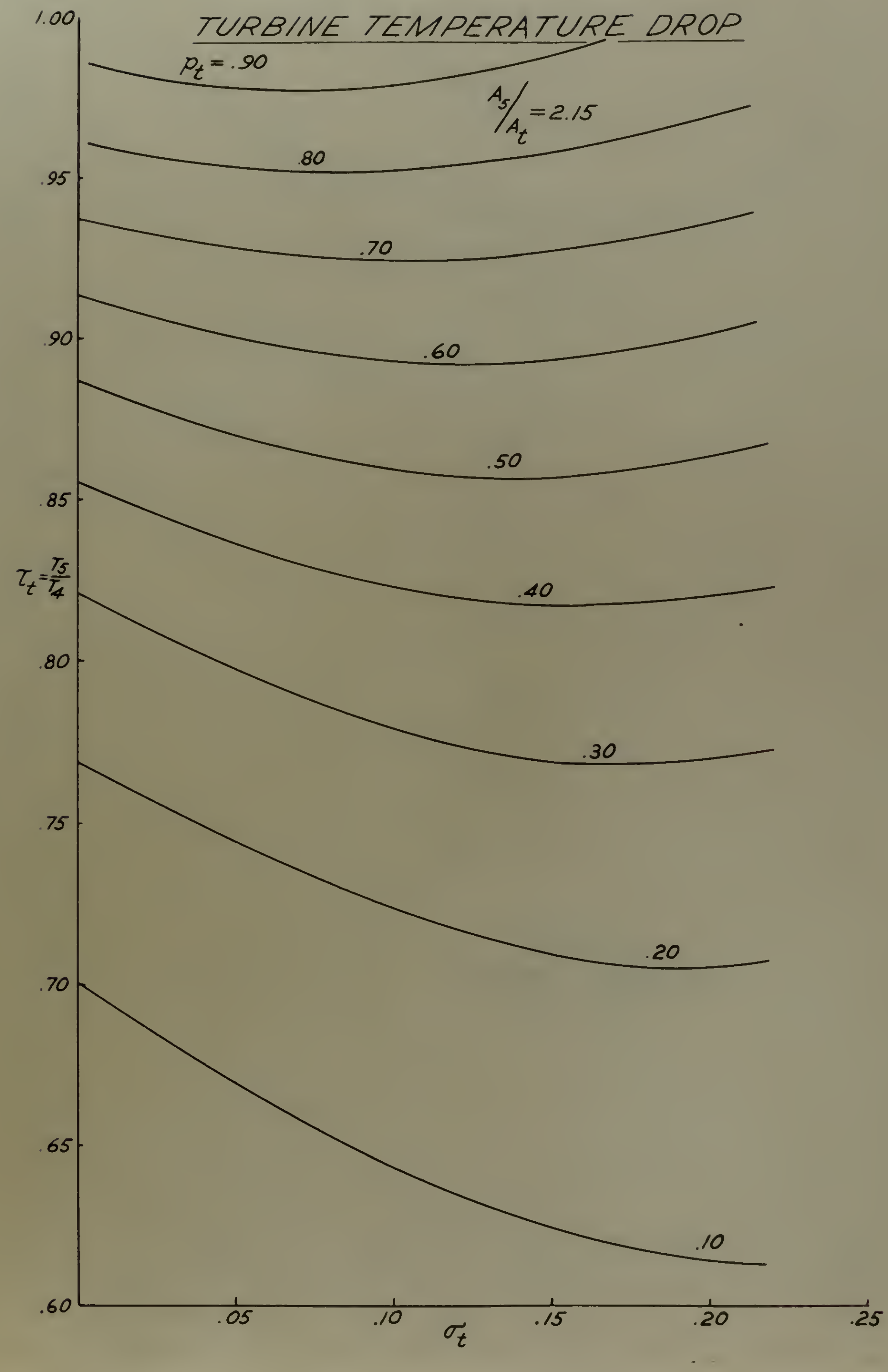
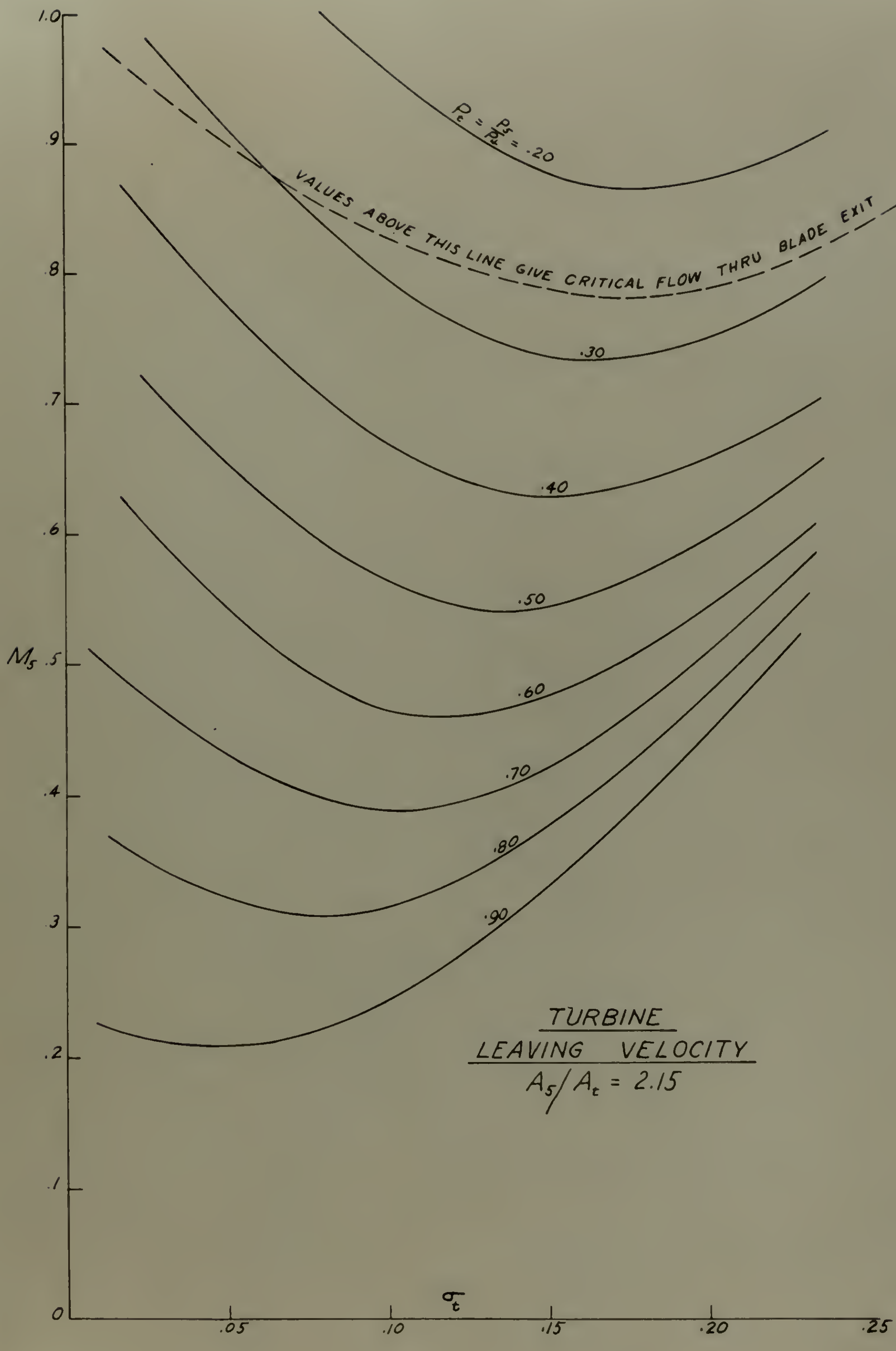
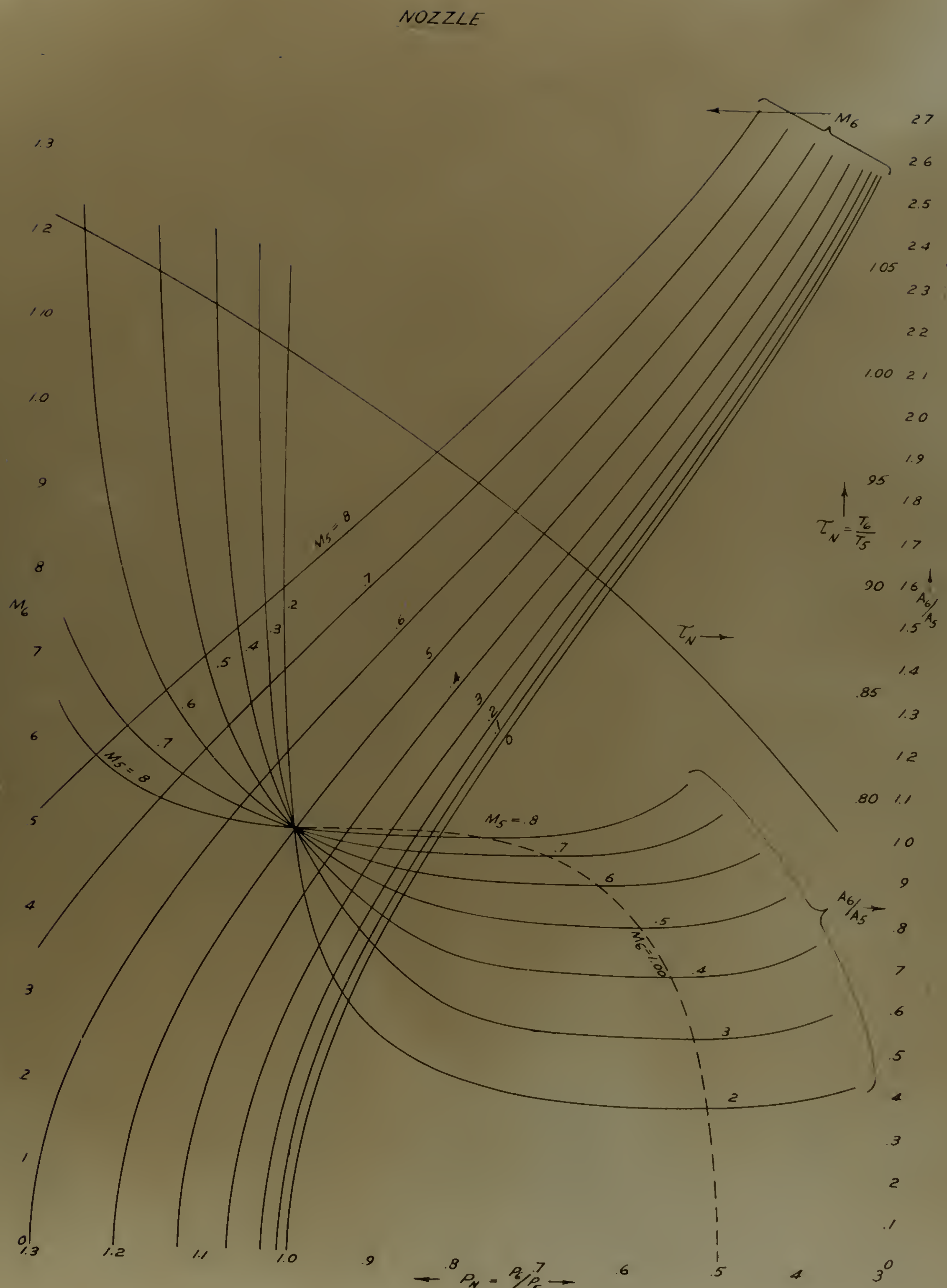

TURBINE TEMPERATURE DROP
$p_t = .90$
$A_5/A_t = 2.15$
.80
.70
.60
.50
.40
.30
.20
.10
$\tau_t = \frac{T_5}{T_4}$
1.00
.95
.90
.85
.80
.75
.70
.65
.60
.05
.10
.15
.20
.25
σ_t

FIGURE 19



1.0
.9
.8
.7
.6
M_5 .5
.4
.3
.2
.1
0
.05
.10
.15
.20
.25
σ_t
$P_t = \frac{P_5}{P_4} = .20$
.30
.40
.50
.60
.70
.80
.90
VALUES ABOVE THIS LINE GIVE CRITICAL FLOW THRU BLADE EXIT
TURBINE
LEAVING VELOCITY
$A_5/A_t = 2.15$

FIGURE 20



APPENDIX I

CALCULATIONS

	LINE	ITEM	1	2	3	4	5	6	7
	1	$\dot{m}$	0.18	0.14	0.15	0.16	0.17	0.19	0.20
*	2	n	100	100	100	100	100	100	100
	3	P_2	2133	2137	2140	2137	2135	2132	2130
	4	T_2	520	520	520	520	520	520	520
	5	$\sqrt{T_2}$	22.8	–	–	–	–	–	–
	6	M_2	0.082	0.064					
	7	Γ_c	0.0686	0.0535					
	8	σ_c	.0935	.0935	.0935	.0935	.0935	.0935	.0935
	9	P_3/P_2	1.16	1.17					
	10	P_3	2470	2500					
	11	Ω_c	0.02	0.02					
	12	W_c	38500	38500	38500	38500	38500	38500	38500
	13	T_3/T_2	1.053	1.055					
	14	T_3	547	549	549	549	548	548	548
	15	P_4	2420	2450	2440	2430	2425	2415	2410
*	16	T_4	2000	2000	2000	2000	2000	2000	2000
	17	$\sqrt{T_4}$	44.7	–	–	–	–	–	–
	18	Γ_t	.239	.184	.198	.212	.225	.253	.268
	19	σ_t	.0515	.0515	.0515	.0515	.0515	.0515	.0515
	20	P_5/P_4	.895	.930	.923	.913	.908	.884	.870
	21	P_5	2165	2280	2250	2220	2200	2130	2100
	22	Ω_t	.017	.010	.010	.012	.015	.020	.023
	23	W_t	35600	21300	21100	25200	31500	42000	47900
	24	T_5/T_4	.977	.985	.988	.985	.982	.976	.971
	25	T_5	1954	1970	1976	1970	1964	1952	1942
	26	M_5	.21	.18	.19	.20	.20	.22	.24
	27	P_6/P_5	.975	.927	.940	.952	.962	.993	1.01
*	28	A_6/A_5	.78	.50	.60	.67	.69	.92	1.04
	29	M_6	.29	.32	.33	.33	.33	.25	.23
	30	T_6/T_5	.99	.98	.985	.982	.98	.995	1.00
	31	T_6	1933	1932	1950	1935	1925	1945	1942
	32	a_6	2095	2095	2100	2095	2090	2100	2100
	33	V_6	608	671	693	692	690	526	483
	34	F	83.0	73.5	82	87	92	72	69
	35	W_t	35600	21300	21100	25200	31500	42000	47900
	36	W_c	38500	38500	38500	38500	38600	38500	38500
	37	ΔW	−2900	−17200	−17400	−13300	−7000	+3500	+9400
*	38	$\dot{h}$	−1.0	−5.6	−5.7	−4.4	−2.3	+1.1	+3.1
	39			Stall					

CALCULATIONS

	8	9	10	11	12	13	14	15	16
1	0.21	0.215	0.34	0.35	0.36	0.37	0.38	0.51	0.52
*2	100	100	150	150	150	150	150	200	200
3	2130	2130	2102	2100	2098	2095	2092	2050	2047
4	520	520	517	517	517	516	516	513	513
5	22.8	22.8	22.7	22.7	22.7	22.7	22.7	22.65	22.65
6			.156	.160	.165	.170	.175	.239	.245
7			.1306	.134	.138	.142	.146	.200	.205
8	.0935	.0935	.141	.141	.141	.141	.141	.188	.188
9			1.45	1.45	1.44	1.44	1.43	1.90	1.89
10			3050	3045	3020	3020	2995	3890	3860
11	0.02	0.02	0.10	0.10	0.10	0.10	0.10	.260	.258
12	38500	38500	189000	189000	189000	188000	188000	477000	473000
13			1.13	1.13	1.13	1.13	1.13	1.240	1.237
14	547	547	583	583	583	582	582	637	636
15	2400	2398	2990	2985	2960	2960	2935	2815	3785
*16	2000	2000	2000	2000	2000	2000	2000	2000	2000
17	44.7	-	-	-	-	-	-	-	-
18	.281	.289	.366	.377	.391	.402	.417	.430	.442
19	.0515	.0515	.0771	.0771	.0771	.0771	.0771	.103	.103
20	.862	.848	.735	.721	.688	.656	.628	.575	.528
21	2070	2035	2200	2150	2040	1940	1840	2190	2000
22	.027	.029	.071	.077	.087	.101	.112	.145	.168
23	56100	60200	184000	199000	223000	259000	287000	479000	551000
24	.966	.965	.936	.921	.922	.912	.903	.885	.869
25	1932	1920	1870	1860	1845	1820	1806	1770	1740
26	.26	.27	.38	.39	.42	.46	.48	.49	.535
27	1.02	1.04	.960	.983	1.037	1.09	1.15	.965	1.057
*28	1.18	2.10	.88	.96	1.12	1.43	2.0	.91	1.12
29	.22	.13	.45	.43	.36	.31	.23	.53	.46
30	1.005	1.007	.985	.992	1.01	1.02	1.035	.986	1.0104
31	1932	1930	1840	1840	1860	1855	1865	1745	1765
32	2100	2100	2045	2055	2055	2050	2055	1990	2000
33	462	273	920	878	741	636	473	1053	920
34	66	27	253	255	214	181	124	462	402
35	56100	60200	184000	199000	223000	259000	287000	479000	551000
36	38500	38500	189000	189000	189000	189000	189000	477000	473000
37	17600	21700	-5000	+10000	+34000	+70000	+98000	+2000	+78000
*38	+5.8	+7.2	-1.1	+2.2	+7.5	+15.4	+21.6	+0.3	+12.9
39			Stall					Stall	

CALCULATIONS

	17	18	19	20	21	22	23	24	25
1	0.53	0.54	0.55	0.545	0.595	0.61	0.62	0.628	0.60
*2	200	200	200	200	225	225	225	225	225
3	2045	2040	2035	2035	2020	2016	2010	2005	2020
4	513	513	513	512	511	511	510	510	511
5	22.65	22.65	22.65	22.6	22.6	22.6	22.55	22.55	22.6
6	.25	.255	.260	.257	.238	.245	.249	.253	.240
7	.209	.213	.217	.215	.212	.212	.212	.212	.212
8	.188	.188	.188	.188	.284	.293	.298	.303	.307
9	1.88	1.87	1.84	1.86	2.20	2.17	2.155	2.13	2.195
10	3840	3825	3755	3780	4440	4380	4340	4265	4430
11	.256	.254	.252	.253	.370	.366	.362	.359	.369
12	468,—	464,—	460,—	462,—	668000	658,—	649,—	641,—	665,—
13	1.234	1.231	1.229	1.230	1.357	1.344	1.332	1.320	1.352
14	634	633	632	632	692	685	680	673	690
15	3765	3750	3680	3705	4350	4290	4250	4180	4340
*16	2000	2000	2000	2000	2000	2000	2000	2000	2000
17	44.7	–	–	–	–	–	–	–	–
18	.453	.463	.480	.473	.440	.457	.469	.482	.444
19	.103	.103	.103	.103	.116	.116	.116	.116	.116
20	.490	.445	.360	.400	.530	.470	.405	.343	.510
21	1830	1670	1325	1480	2390	2020	1720	1430	2215
22	.190	.212	.258	.234	.173	.207	.244	.278	.183
23	619,—	690,—	823,—	750,—	652,—	769,—	898,—	1007,—	705,—
24	.857	.842	.807	.824	.870	.848	.825	.799	.861
25	1705	1685	1615	1650	1740	1696	1650	1598	1722
26	.575	.616	.710	.670	.525	.58	.642	.719	.64
27	1.155	1.267	1.60	1.36	.885	1.05	1.23	1.52	.955
*28	1.37	1.98	This	3.0	.88	1.075	1.49	very high	0.94
29	0.36	0.23	mass flow	0.10	.67	.52	.38	(×10)	.60
30	1.036	1.06	not	1.08	.967	1.012	1.055		.987
31	1765	1785	reached when	1785	1683	1715	1740		1700
32	2000	2010	$A_{V_2} \to \infty$	2010	1950	1970	1990		1960
33	720	463	m_s	201	1305	1025	757		1175
34	304	171		5	690	537	378		617
35	619,—	690,—	823,—	750,—	652,—	769,—	898,—	1007,—	705,—
36	468,—	464,—	460,—	462,—	668,—	658,—	649,—	641,—	665,—
37	+151,—	+226,—	+363,—	+288,—	−16,—	+11,000	+249,—	+366,—	+40,—
*38	+24.9	+37.2	+59.9	+47.5	−2.3	+16.2	+36.5	+53.5	+5.9
39					Stall			Choke	

CALCULATIONS

	26	27	28	29	30	31	32	33
1	0.615	0.623	0.68	0.69	0.70	0.71	0.715	0.72
* 2	225	225	250	250	250	250	250	250
3	2010	2005	1980	1974	1967	1960	1958	1954
4	510	510	508	508	508	507	507	507
5	22.55	22.55	22.5	22.5	22.5	22.5	22.5	22.5
6	.295	.300	.331	.340	.343	.349	.352	.355
7	.247	.251	.277	.282	.287	.292	.294	.297
8	.212	.212	.236	.236	.236	.236	.236	.236
9	2.165	2.145	2.60	2.59	2.575	2.56	2.55	2.54
10	4380	4330	5150	5110	5070	5020	4990	4970
11	.364	.361	.515	.515	.515	.515	.513	.512
12	652,—	645,—	908,000	905,000	903,000	901,000	896,000	891,000
13			1.425	1.421	1.417	1.413	1.411	1.409
14	682	678						
15	4290	4240	5050	5010	4970	4920	4890	4870
* 16	2000	2000	2000	2000	2000	2000	2000	2000
17	44.7	✓	✓	✓	✓	✓	✓	✓
18	.461	.472	.433	.442	.453	.463	.470	.475
19	.116	.116	.128	.128	.128	.128	.128	.128
20	.438	.395	.549	.512	.479	.431	.400	.372
21	1880	1675	2770	2565	2375	2120	1955	1810
22	.225	.253	.162	.182	.203	.235	.252	.272
23	837,—	930,—	708,—	790,—	875,—	1003,—	1070,—	1147,—
24	.835	.813	.874	.862	.848	.831	.819	.806
25	1670	1626	1748	1724	1696	1662	1638	1612
26	.608	.66	.50	.535	.57	.61	.64	.67
27	1.125	1.265	.765	.826	.893	1.00	1.085	1.17
* 28	1.21	1.63	.80	.84	.91	1.00	1.08	1.23
29	.46	.33	.79	.74	.69	.61	.56	.48
30	1.029	1.06	.934	.953	.97	1.00	1.02	1.04
31	1720	1725	1635	1645	1645	1662	1670	1680
32	1970	1970	1920	1930	1930	1940	1940	1950
33	907	651	1517	1430	1330	1180	1090	937
34	468	314	932	885	828	733	675	568
35	837,000	930,000	708,—	790,—	875,—	1003,—	1070,—	1147,—
36	652,000	645,000	908,—	905,—	903,—	901,—	896,—	891,—
37	+185,000	+285,000	-200,—	-115,—	-28,—	+102,—	+174,—	+256,—
* 38	+27.1	+41.8	-26.4	-15.2	-3.7	+13.5	+23.0	+33.8
39			Stall					

CALCULATIONS

	34	35	36	37	38	39	40	41	42
1	0.725	0.755	0.77	0.78	0.79	0.80	0.805	0.81	0.815
*2	250	275	275	275	275	275	275	275	275
3	1951	1935	1925	1918	1911	1903	1899	1895	1891
4	505	505	504	504	504	503	503	502	502
5	22.5	22.45	22.45	22.45	22.45	22.45	22.45	22.4	22.4
6	.358	.377	.387	.394	.399	.405	.410	.413	.417
7	.299	.316	.324	.330	.334	.339	.343	.346	.348
8	.266	.261	.261	.261	.261	.261	.261	.261	.261
9	2.525	3.11	3.10	3.09	3.07	3.03	2.99	2.96	2.93
10	4930	6020	5970	5920	5870	5770	5680	5610	5550
11	.509	.715	.714	.713	.711	.709	.707	.705	.703
12	885, —	1230, —	1220, —	1215, —	1208, —	1200, —	1191, —	1187, —	1184, —
13	1.407	1.51	1.50	1.49	1.48	1.47	1.46	1.45	1.44
14									
15	4835	5900	5850	5800	5760	5655	5565	5500	5440
*16	2000	2000	2000	2000	2000	2000	2000	2000	2000
17	44.7	✓	✓	—	—	—	—	—	—
18	.482	.411	.422	.433	.442	.454	.465	.473	.482
19	.128	.141	.141	.141	.141	.141	.141	.141	.141
20	.333	.608	.566	.542	.503	.471	.418	.382	.323
21	1610	3585	3310	3140	2890	2660	2325	2100	1760
22	.298	.123	.151	.164	.184	.207	.244	.264	.310
23	1250, —	628, —	765, —	824, —	917, —	1016, —	1178, —	1258, —	1460, —
24	.789	.895	.880	.872	.858	.844	.825	.809	.782
25	1578	1790	1760	1745	1716	1688	1650	1618	1564
26	.72	.46	.50	.52	.54	.57	.61	.66	.71
27	1.315	0.59	.639	.673	.732	.795	.908	1.008	1.203
*28	1.62	.74	.78	.81	.83	.87	.93	1.01	1.22
29	.36	.98	.94	.93	.85	.80	.71	.65	.48
30	1.07	.88	.90	.907	.928	.943	.975	1.03	1.047
31	1690	1575	1585	1580	1595	1590	1610	1665	1640
32	1960	1890	1895	1890	1905	1900	1910	1940	1930
33	706	1855	1785	1755	1620	1520	1357	1260	927
34	406	1290	1260	1255	1165	1100	975	903	636
35	1250, —	1230, —	765, —	824, —	917, —	1016, —	1178, —	1258, —	1460, —
36	885, —	628, —	1220, —	1215, —	1208, —	1200, —	1191, —	1187, —	1184, —
37	+365, —	-607, —	-455, —	-391, —	-291, —	-184, —	-13, —	+71, —	+276, —
*38	+48.2	-72.3	-54.7	-47.0	-35.0	-22.1	-1.6	+8.5	+33.2
39	Choke	Stall							Choke

CALCULATIONS

	43	44	45	46	47	48	49	50	51
1	0.83	0.84	0.85	0.86	0.87	0.88	0.89	0.897	0.708
★ 2	300	300	300	300	300	300	300	300	245
3	1880	1875	1870	1860	1850	1840	1830	1825	1960
4	501	501	500	500	499	498	498	498	507
5	22.35	22.35	22.35	22.35	22.3	22.3	22.3	22.3	22.5
6	.428	.435	.443	.450	.457	.464	.471		.348
7	.358	.364	.371	.377	.383	.388	.394		.291
8	.286	.286	.286	.286	.286	.286	.286	.286	.2315
9	3.75	3.73	3.71	3.67	3.61	3.54	3.44	3.35	2.435
10	7050	7000	6930	6820	6690	6510	6300	6120	4770
11	.975	.970	.965	.959	.951	.940	.934	.930	
12	1620,—	1610,—	1595,—	1580,—	1555,—	1530,—	1510,—	1505,—	
13	1.613	1.599	1.585	1.572	1.559	1.546	1.530	1.518	
14	808	801	792	786	778	771			
15	6910	6860	6790	6680	6560	6380	6180	6000	4675
★ 16	2000	2000	2000	2000	2000	2000	2000	2000	2000
17	44.7	–	–	–	–	–	–	–	–
18	.385	.393	.402	.414	.427	.443	.463	.482	.486
19	.154	.154	.154	.154	.154	.154	.164	.154	
20	.663	.642	.620	.593	.556	.508	.409	.322	Above
21	4580	4410	4210	3980	3840	3240	2530	1930	Critical
22	.090	.100	.113	.129	.150	.179	.248	.312	Flow.
23	539,—	595,—	665,—	747,—	852,—	989,—	1327,—	1620,—	
24	.914	.907	.901	.880	.867	.859	.822	.780	
25	1828	1814	1802	1760	1734	1718	1644	1560	
26	.45	.459	.470	.486	.51	.54	.625	.71	
27	.461	.479	.502	.531	.581	.653	.836	1.095	
★ 28	.74	.75	.76	.78	.80	.82	.91	1.06	
29	1.00	1.00	1.00	1.00	1.00	0.96	0.84	0.62	
30	For thrust computation					0.90	.955	1.022	
31	when $M_0 = 1.0$ see Appendix II					1545	1570	1593	
32						1880	1890	1900	
33						1805	1590	1178	
34	1795	1755	1710	1640	1555	1460	1285	925	
35	539,—	595,—	665,—	747,—	852,—	989,—	1327,—	1620,—	
36	1620,—	1610,—	1595,—	1580,—	1555,—	1530,—	1510,—	1505,—	
37	-1080,—	-1015,—	-930,—	-833,—	-703,—	-541,—	-183,—	+115,—	
★ 38	-179	-111	-102	-92	-77	-60	-20.2	+12.6	
39	Stall							Choke	

CALCULATIONS

	52	53	54	55	56	57	58	59	60
1	0.706	0.774	0.926	0.14	0.16	0.18	0.20	0.21	0.34
*2	245	263	310	100	100	100	100	100	150
3	1960	1923	1800						
4	507	504	495						
5	22.5	22.45	22.25			See items 2-8 incl.			
6	.346	.390	.498						
7	.2895	.326	.417						
8	2.445	2.75	3.50						
9	.2315	.249	.296						
10	4805	5280							
11	0.470	.605	.981						
12	818,000	1033000	1555000	38500	38500	38500	38500	38500	189000
13									
14				549	549	549	549	549	583
15	4703	5150	6170	2450	2430	2420	2410	2400	2990
*16	2000	2000	2000	2200	2200	2200	2200	2200	2200
17	44.7	✓	✓	46.9	✓	✓	✓	✓	✓
18	.482	.482	.482	.193	.223	.251	.2805	.295	.384
19	.126	.135	.159	.049	.049	.049	.049	.049	.0735
20	.335	.324	.314	.930	.908	.887	.862	.842	.704
21	1575	1670	1938	2280	2210	2145	2075	2020	2105
22	.293	.308	.319	.012	.015	.019	.025	.031	.082
23	1195000	1373000	1705000	26800	32100	41800	54800	67700	223500
24				.984	.979	.975	.969	.965	.926
25				2165	2155	2150	2130	2120	2040
26	.72	.73	.72	.14	.20	.23	.25	.27	.40
27	1.343	1.267	1.092	.937	.958	.986	1.02	1.048	1.003
*28	1.79	1.38	1.06	.43	.68	.89	1.23	2.05	1.006
29				.36	.32	.29	.22	.13	.40
30				.979	.988	.996	1.005	1.012	1.002
31				2120	2130	2140	2145	2145	2045
32				2190	2200	2205	2210	2210	2150
33				788	703	640	487	287	861
34				90	89	89	68	29	243
35	1195 —	1373 —	1705 —	26800	32100	41800	54800	67700	223500
36	818 —	1033 —	1555 —	38500	38500	38500	38500	38500	189000
37	377 —	340 —	150 —	−11700	−6400	+3300	+16300	+29300	+34500
*38	+57.7	+42.5	+16.0	−3.5	−2.1	+1.1	+5.4	+9.7	+7.6
39	Choke	Choke	Choke	Stall					Stall

CALCULATIONS

1	61	62	63	64	65	66	67	68	69
1	0.35	0.36	0.37	0.52	0.515	0.51	0.595	0.60	0.61
*2	150	150	150	200	200	200	225	225	225
3						2052			
4						513			
5						22.65			
6						.2385			
7						.199			
8						.188			
9						1.885			
10						3865			
11						.248			
12	189,—	189,—	188,—	473,—	475,—	457,—	668,—	665,—	658,—
13									
14	583	583	582	636	636	637	692	690	685
15	2985	2960	2950	3785	3800	3788	4350	4340	4290
*16	2200	2200	2200	2200	2200	2200	2200	2200	2200
17	46.9	✓	✓	✓	✓	✓	✓	✓	✓
18	.396	.411	.423	.463	.458	.454	.461	.465	.478
19	.0735	.0735	.0735	.0981	.0981	.0981	.110	.110	.110
20	.679	.643	.602	.432	.461	.484	.437	.414	.376
21	2030	1905	1775	1633	1750	1835	1900	1795	1650
22	.091	.103	.118	.219	.203	.192	.223	.232	.258
23	237,500	277,—	317,—	755,—	702,—	662,—	883,—	915,—	1008,—
24	.919	.909	.897	.836	.846	.854	.836	.827	.812
25	2025	2000	1975	1840	1865	1880	1840	1820	1790
26	.43	.46	.50	.63	.60	.58	.62	.64	.69
27	1.043	1.11	1.192	1.297	1.21	1.154	1.113	1.18	1.283
*28	1.17	1.53	3.0	2.5	1.58	1.37	1.20	1.31	1.54
29	.36	.28	.11	.20	.32	.38	.48	.43	.36
30	1.01	1.025	1.045	1.06	1.049	1.036	1.026	1.042	1.06
31	2045	2050	2060	1950	1955	1950	1890	1900	1900
32	2150	2155	2160	2110	2110	2110	2075	2080	2080
33	774	603	238	422	676	803	996	895	746
34	219	164	33	143	272	334	506	449	366
35	237500	277,—	317,—	755,—	702,—	662,—	883,—	915,—	1008,—
36	189,—	189,—	188,—	473,—	475,—	457,—	668,—	665,—	658,—
37	48500	88000	129,—	282,—	227,—	205,—	215,—	250,—	350,—
*38	+10.7	19.4	+28.4	+46.5	+37.5	+33.8	+31.5	+36.7	+51.3
39						Stall	Stall		

CALCULATIONS

	70	71	72	73	74	75	76	77	78
1	0.612	0.611	0.68	0.685	0.69	0.695	0.70	0.705	0.755
★ 2	225	225	250	250	250	250	250	250	275
3									
4									
5									
6									
7									
8									
9									
10									
11									
12	656000	657,—	908,—	906,—	905,—	904,—	903,—	902,—	1,230,000
13									
14	684	684	5050	5030	5020	4990	4970	4945	758
15	4280	4285							5900
★ 16	2200	2200	2200	2200	2200	2200	2200	2200	2200
17	46.9	46.9	✓	✓	✓	✓	✓	✓	✓
18	0.482	0.480	.455	.459	.464	.470	.476	.482	.431
19	.342	.359	.122	.122	.122	.122	.122	.122	.135
20	1.465	1.537	.467	.449	.421	.395	.366	.335	.541
21	.275		2360	2260	2110	1975	1820	1655	3090
22			.204	.221	.237	.252	.270	.288	.166
23	1,072—	1040,—	937,—	1013,—	1082,—	1145,—	1220,—	1297,—	882,000
24	.796	.805	.844	.838	.828	.817	.804	.789	.871
25	1755	1770	1860	1845	1825	1800	1770	1735	1945
26	.73	.71	.578	.596	.624	.649	.682	.721	.51
27	1.447	1.375	.896	.935	1.003	1.07	1.163	1.28	.683
★ 28	4½	2.1	.91	.94	1.003	1.07	1.19	1.44	.80
29	0+	0.26	.71	.67	.62	.58	.515	.42	.91
30	1.1	1.075	.971	.982	1.001	1.016	1.038	1.065	.91
31	1930	1905	1807	1815	1840	1830	1835	1850	1740
32	2090	2080	2025	2030	2050	2040	2040	2050	1990
33	0	541	1435	1360	1270	1183	1050	861	1810
34	0	240	877	832	775	720	633	503	1250
35	1072—	1040—	937—	1013,—	1082,—	1145—	1220—	1297—	882,000
36	656—	657,—	908,—	906,—	905,—	904—	903,—	902,—	1230,000
37	416,—	383,—	+27,—	107,—	177,—	241,—	317,—	395,—	-348,000
★ 38	+61.2	+56.1	+3.8	+14.1	+23.4	+31.8	+41.9	+52.1	-41.8
39	Choke		Stall					Choke	Stall

CALCULATIONS

	79	80	81	82	83	84	85	86	87
1	0.77	0.78	0.79	0.80	0.802	0.83	0.85	0.87	0.88
* 2	275	275	275	275	275	300	300	300	300
3									
4									
5									
6									
7									
8									
9									
10									
11									
12	1220,000	1215,—	1208,—	1200,—	1198,—	1620,—	1595,—	1555,—	1530,—
13									
14	757	755	754	752	752	808	792	778	771
15	5850	5800	5750	5655	5620	6910	6790	6550	6380
* 16	2200	2200	2200	2200	2200	2200	2200	2200	2200
17	46.9	—	—	—	—	—	—	—	—
18	.442	.455	.464	.476	.482	.404	.421	.448	.465
19	.135	.135	.135	.135	.135	.147	.147	.147	.147
20	.502	.456	.418	.362	.328	.620	.562	.478	.410
21	2935	2640	2405	2045	1845	4280	3820	3135	2620
22	.188	.218	.242	.282	.300	.114	.160	.202	.248
23	1,003,000	1153,—	1265,—	1452,—	1533,—	718,—	928,—	1208,—	1443,—
24	.857	.839	.825	.801	.785	.899	.878	.851	.822
25	1885	1845	1815	1763	1730	1980	1935	1870	1810
26	.54	.58	.62	.68	.72	.46	.50	.57	.62
27	.722	.802	.88	1.034	1.147			.674	.807
* 28	.83	.88	.93	1.02	1.12	See		.85	.91
29	.875	.80	.765	.65	.58	Appendix II		.75	.83
30	.929	.946	.966	1.009	1.034			.71	.947
31	1750	1747	1755	1780	1785			1702	1717
32	1995	1995	1995	2010	2010			1970	1980
33	1745	1595	1527	1307	11[illegible]7			1875	1645
34	1230	1130	1090	933	818			1505	1245
35	1003,000	1153,—	1265,—	1452,—	1533,—	718,—	928,—	1208,—	1443,—
36	1220,000	1215,—	1208,—	1200,—	1198,—	1620,—	1595,—	1555,—	1530,—
37	−217,000	−62,—	+57,—	252,—	+335,—	−902,—	−667,—	−347,—	−87,—
* 38	−26.0	−7.4	+6.9	+30.2	+40.2	−99.3	−73.4	−38.2	−9.6
39					Chote				

CALCULATIONS

	88	89	90	91
1	0.888	0.669	0.757	0.859
*2	300	240	262	290
3		1980	1935	1865
4		508	505	500
5		22.5	22.4	22.3
6		.32	.375	.445
7		.268	.314	.372
8		.227	.249	.278
9		.242	.281	3.31
10		4790	5430	6170
11		.459	.604	.831
12	1515000	812,000	1037,000	1,370,000
13				
14	763			
15	6220	4700	5320	6050
*16	2200	✓	✓	✓
17	46.9	✓	✓	✓
18	.482	.482	.482	.482
19	.147	.118	.128	.142
20	.320	.341	.33	.325
21	1990	1605	1755	1965
22	.313	.284	.294	.306
23	1775,—	1215,—	1423,—	1680,—
24	.780	.793	.787	.782
25	1717	1743	1730	1720
26	.72	.732	.717	.713
27	1.063	1.318	1.206	1.078
*28	1.03	1.60	1.21	1.04
29	.65	.39	.495	.64
30	1.015	1.07	1.048	1.018
31	1740	1868	1812	1750
32	1990	2060	2030	1990
33	1295	803	1006	1272
34	1020	438	642	967
35	1775,000	1,215,000	1423,—	1680,—
36	1515,000	812,—	1037,—	1370,—
37	+260,000	403,—	386,—	310,—
*38	+28.6	+55.4	+48.0	+35.2
39	Choke	Choke	Choke	Choke

APPENDIX II

THRUST COMPUTATION WHEN $\frac{P_{55}}{P_0} > P_{crit}$

$$F = \dot{m}(V_6 - V_0) + A_6(P_6 - P_0)$$

$$F = \dot{m}(V_6 - V_0) + A_5\left(\frac{A_6}{A_5}\right)\left[P_5\left(\frac{P_6}{P_5}\right) - P_0\right]$$

Find $\frac{P_6}{P_5}$ from Nozzle Chart using given M_5, and $M_6 = 1.0$

$\frac{A_6}{A_5}$ from Nozzle Chart using given M_5 and $M_6 = 1.0$

$A_5 = 0.877\ ft^2$

	$n = 300$ $T_4 = 2000°$					$n = 300$ $T_4 = 2200°$	
$\dot{m}$	0.83	0.84	0.85	0.86	0.87	0.83	0.85
M_5	.45	.459	.47	.486	.51	.46	.50
P_5	4580	4410	4210	3980	3640	4280	3820
M_6	1.0	1.0	1.0	1.0	1.0	1.0	1.0
P_6/P_5	.573	.578	.583	.590	.603	.578	.600
P_6	2630	2550	2460	2340	2200	2480	2290
$P_6 - P_0$	514	434	344	224	084	464	174
A_6/A_5	.74	.75	.75	.76	.80	.75	.78
T_5	1828	1814	1802	1760	1734	1980	1935
T_6/T_5	.875	.876	.877	.878	.880	.876	.880
T_6	1600	1588	1580	1543	1528	1735	1700
$V_6 = a_6$	1905	1900	1895	1870	1865	1980	1970
$m(V_6 - V_0)$	1460	1470	1485	1490	1495	1523	1513
$A_6(P_6 - P_0)$	333	285	226	148	59	305	119
F	1795	1755	1711	1638	1554	1828	1632

APPENDIX III

DETERMINATION OF MEAN GAMMA

$$\gamma = \frac{c_p}{c_v} = \frac{c_p}{c_p - R}$$

Compressor: Mean T = 620°R
Fluid : Air

$$\gamma = \frac{6020}{6020-1718} = 1.40$$

Turbine: Inlet T 1500°R to 2400°R
Outlet T 1200°R to 2000°R
Average T 1350°R to 2200°R

Fluid : Fuel/Air Ratio 0.01 to 0.03

Temp.	Fuel/Air	c_p	R	γ
1350°R	0.01	6570	1716	1.35
1350°R	0.03	6750	1716	1.34
2200°R	0.01	7390	1710	1.30
2200°R	0.03	7110	1716	1.31
			Avg. =	1.33

Note: Values of c_p and R obtained from Hinkel and Turner, NACA ARR 4B25

Thesis 8109
S18 Satterfield
Theoretical investigation of acceleration of a turbojet engine.